CHAPTER 12
AIRCRAFT INSTRUMENT SYSTEMS

GENERAL

Safe, economical, and reliable operation of modern aircraft is dependent upon the use of instruments. The first aircraft instruments were fuel and oil pressure instruments to warn of engine trouble so that the aircraft could be landed before the engine failed. As aircraft that could fly over considerable distances were developed, weather became a problem. Instruments were developed that helped to fly through bad weather conditions.

Instrumentation is basically the science of measurement. Speed, distance, altitude, attitude, direction, temperature, pressure, and r.p.m. are measured and these measurements are displayed on dials in the cockpit.

There are two ways of grouping aircraft instruments. One is according to the job they perform. Within this grouping they can be classed as flight instruments, engine instruments, and navigation instruments. The other method of grouping aircraft instruments is according to the principle on which they work. Some operate in relation to changes in temperature or air pressure and some by fluid pressure. Others are activated by magnetism and electricity, and others depend on gyroscopic action.

The instruments that aid in controlling the in-flight attitude of the aircraft are known as flight instruments. Since these instruments must provide information instantaneously, they are located on the main instrument panel within ready visual reference of the pilot. Basic flight instruments in an aircraft are the airspeed indicator, altimeter and the magnetic direction indicator. In addition, some aircraft may have a rate-of-turn indicator, a bank indicator, and an artificial horizon indicator. Flight instruments are operated by atmospheric, impact, differential, or static pressure or by a gyroscope.

Engine instruments are designed to measure the quantity and pressure of liquids (fuel and oil) and gases (manifold pressure), r.p.m., and temperature. The engine instruments usually include a tachometer, fuel and oil pressure gages, oil temperature gage, and a fuel quantity gage. In addition some aircraft that are powered by reciprocating engines are equipped with manifold pressure gage(s), cylinder head temperature gage(s), and carburetor air-temperature gage(s). Gas turbine powered aircraft will have a turbine or tailpipe temperature gage(s), and may have an exhaust pressure ratio indicator(s).

Navigational instruments provide information that enables the pilot to guide the aircraft accurately along definite courses. This group of instruments includes a clock, compasses (magnetic compass and gyroscopic directional indicator), radios, and other instruments for presenting navigational information to the pilot.

INSTRUMENT CASES

A typical instrument can be compared to a clock, in that the instrument has a mechanism, or works; a dial, or face; pointers, or hands; and a cover glass. The instrument mechanism is protetced by a one; or two-piece case. Various materials, such as aluminum alloy, magnesium alloy, iron, steel, or plastic are used in the manufacture of instrument cases. Bakelite is the most commonly used plastic. Cases for electrically operated instruments are made of iron or steel; these materials provide a path for stray magnetic force fields that would otherwise interfere with radio and electronic devices.

Some instrument mechanisms are housed in airtight cases, while other cases have a vent hole. The vent allows air pressure inside the instrument case to vary with the aircraft's change in altitude.

DIALS

Numerals, dial markings, and pointers of instruments are frequently coated with luminous paint. Some instruments are coated with luminous calcium sulphide, a substance that glows for several hours after exposure to light. Other instruments have a phosphor coating that glows only when excited by a small ultraviolet lamp in the cockpit. Some instruments are marked with a combination of radioactive salts, zinc oxide, and shellac. In handling these instruments, care should be taken against radium poisoning. The effects of radium are cumulative and can appear after a long period of continued exposure to small amounts of radiation. Poisoning usually results from touching the mouth or nose after handling instrument dials or radioactive paint. After handling either, the hands should be kept away from the mouth and nose, and washed thoroughly with hot water and soap as soon as possible.

RANGE MARKINGS

Instrument range markings indicate, at a glance, whether a particular system or component is operating in a safe and desirable range of operation or in an unsafe range.

Instruments should be marked and graduated in accordance with the Aircraft Specifications or Type Certificate Data Sheets and the specific aircraft maintenance or flight manual. Instrument markings usually consist of colored decalcomanias or paint applied to the outer edges of the cover glass or over the calibrations on the dial face. The colors generally used as range markings are red, yellow, green, blue, or white. The markings are usually in the form of an arc or a radial line.

A red radial line may be used to indicate maximum and minimum ranges: operations beyond these markings are dangerous and should be avoided. A blue arc marking indicates that operation is permitted under certain conditions. A green arc indicates the normal operating range during continuous operation. Yellow is used to indicate caution.

A white index marker is placed near the bottom of all instruments that have range markings on the cover glass. The index marker is a line extending from the cover glass onto the instrument case. The marker shows if glass slippage has occurred. Glass slippage would cause the range markings to be in error.

INSTRUMENT PANELS

With a few exceptions, instruments are mounted on a panel in the cockpit so that the dials are plainly visible to the pilot or copilot. Instrument panels are usually made of sheet aluminum alloy strong enough to resist flexing. The panels are nonmagnetic and are painted with a nonglare paint to eliminate glare or reflection.

In aircraft equipped with only a few instruments, only one panel is necessary; in some aircraft, additional panels are required. In such cases the forward instrument panel is usually referred to as the "main" instrument panel to distinguish it from additional panels on the cockpit overhead or along the side of the flight compartment. On some aircraft the main instrument panel is also referred to as the pilot's or copilot's panel, since many of the pilot's instruments on the left side of the panel are duplicated on the right side.

The method of mounting instruments on their respective panels depends on the design of the instrument case. In one design, the bezel is flanged in such a manner that the instrument can be flush-mounted in its cutout from the rear of the panel. Integral self-locking nuts are provided at the rear faces of the flange corners to receive mounting screws from the front of the panel. The flanged type case can also be mounted from the front of the panel.

The mounting of instruments that have flangeless cases is a simpler process. The flangeless case is mounted from the front of the panel. A special expanding type of clamp, shaped and dimensioned to fit the instrument case, is secured to the rear face of the panel. As actuating screw is connected to the clamp and is accessible from the front of the panel. The screw can be rotated to loosen the clamp, permitting the instrument to slide freely into the clamp. After the instrument is positioned, the screw is rotated to tighten the clamp around the instrument case.

Instrument panels are usually shock-mounted to absorb low-frequency, high-amplitude shocks. Shock mounts are used in sets of two, each secured to

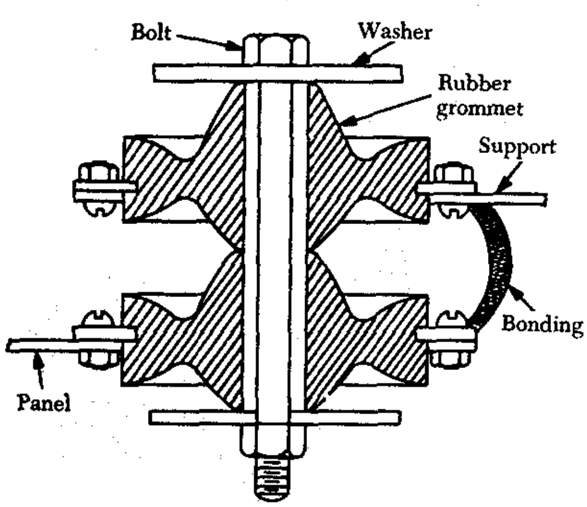


FIGURE 12–1. Section through instrument panel shock.

separate brackets. The two mounts absorb most of the vertical and horizontal vibration, but permit the instruments to operate under conditions of minor vibration. A cross sectional view of a typical shock mount is shown in figure 12-1.

The type and number of shock mounts to be used for instrument panels are determined by the weight of the unit. The weight of the complete unit is divided by the number of suspension points. For example, an instrument panel weighing 16 lbs. which is supported at four points would require eight shock absorbers, each capable of supporting 4 lbs. When the panel is mounted, the weight should deflect the shock absorbers approximately 1/8 in.

Shock-mounted instrument panels should be free to move in all directions and have sufficient clearance to avoid striking the supporting structure. When a panel does not have adequate clearance, inspect the shock mounts for looseness, cracks, or deterioration.

REPAIR OF AIRCRAFT INSTRUMENTS

The repair of aircraft instruments is highly specialized, requiring special tools and equipment. Instrument repairmen must have had specialized training or extensive on-the-job training in instrument repair. For these reasons, the repair of instruments must be performed by a properly certificated instrument repair facility. However, mechanics are responsible for the installation, connection, removal, servicing, and functional checking of the instruments.

AIRCRAFT PRESSURE GAGES

Pressure gages are used to indicate the pressure at which engine oil is forced through the bearings,

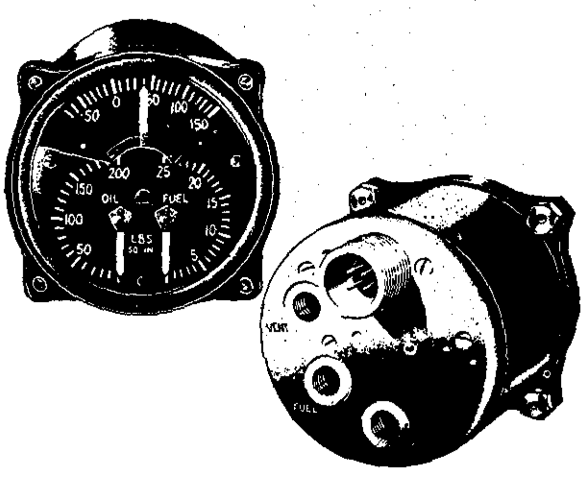


FIGURE 12-2. Engine gage unit.

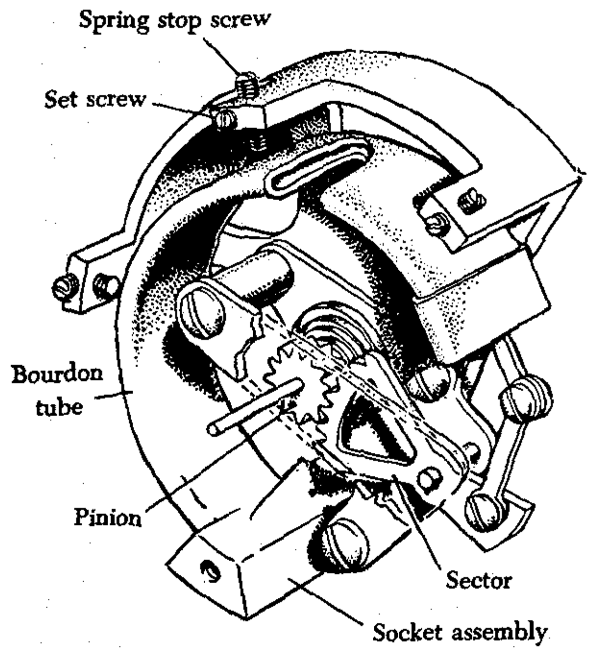


FIGURE 12-3. Bourdon tube pressure gage.

oil passages, and moving parts of the engine and the pressure at which fuel is delivered to the carburetor or fuel control. They are also used to measure the pressure of air in de-icer systems and gyroscope drives, of fuel/air mixtures in the intake manifold, and of liquid or gases in several other systems.

Engine Gage Unit

The engine gage unit is comprised of three separate instruments housed in a single case. A typical engine gage unit, containing gages for oil and fuel pressure and oil temperature, is shown in figure 12-2.

Two types of oil temperature gages are available for use in an engine gage unit. One type consists of an electrical resistance type oil thermometer, supplied electrical current by the aircraft d.c. power system. The other type, a capillary oil thermometer, is a vapor pressure type thermometer consisting of a bulb connected by a capillary tube to a Bourdon tube. A pointer, connected to the Bourdon tube through a multiplying mechanism, indicates on a dial the temperature of the oil.

The Bourdon tube is an aircraft instrument made of metal tubing, oval or somewhat flattened in cross section (figure 12-3). The metal tubing is closed at one end and mounted rigidly in the instrument case at its other end.

The fluid whose pressure is to be measured is introduced into the fixed end of the Bourdon tube by a small tube leading from the fluid system to the instrument. The greater the pressure of the fluid, the more the Bourdon tube tends to become straight. When the pressure is reduced or removed, the inherent springiness of the metal tube causes it to curve back to its normal shape.

If an indicator needle or pointer is attached to the free end of the Bourdon tube, its reactions to changes in the fluid pressure can be observed.

Hydraulic Pressure Gage

The mechanisms used in raising and lowering the landing gear or flaps in most aircraft are operated by a hydraulic system. A pressure gage to measure the differential pressure in the hydraulic system indicates how this system is functioning. Hydraulic pressure gages are designed to indicate either the pressure of the complete system or the pressure of an individual unit in the system.

A typical hydraulic gage is shown in figure 12–4. The case of this gage contains a Bourdon tube and a gear-and-pinion mechanism by which the Bourdon tube's motion is amplified and transferred to the pointer. The position of the pointer on the calibrated dial indicates the hydraulic pressure in p.s.i.

The pumps which supply pressure for the operation of an aircraft's hydraulic units are driven either by the aircraft's engine or by an electric motor, or both. Some installations use a pressure accumulator to maintain a reserve of fluid under pressure at all times. In such cases the pressure gage registers continuously. With other installations, operating pressure is built up only when needed, and pressure registers on the gage only during these periods.

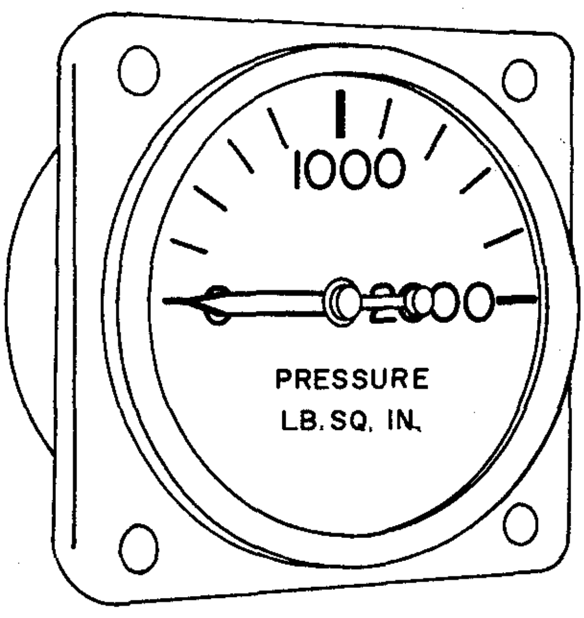


FIGURE 12–4. Hydraulic pressure gage

De-icing Pressure Gage

The rubber expansion boots, which de-ice the leading edges of wings and stabilizers on some aircraft, are operated by a compressed air system. The de-icing system pressure gage measures the difference between prevailing atmospheric pressure and the pressure inside the de-icing system, indicating whether there is sufficient pressure to operate the de-icer boots. The gage also provides a method of measurement when adjusting the relief-valve and the regulator of the de-icing system.

A typical de-icing pressure gage is shown in figure 12–5. The case is vented at the bottom to keep the interior at atmospheric pressure, as well as to provide a drain for any moisture which might accumulate.

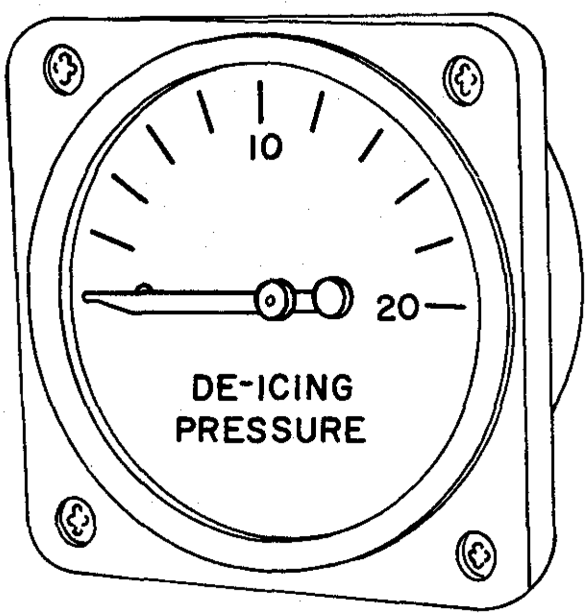


FIGURE 12–5. De-icing pressure gage.

The pressure-measuring mechanism of the de-icing pressure gage consists of a Bourdon tube and a sector gear, with a pinion for amplifying the motion of the tube and transferring it to the pointer. The de-icing system pressure enters the Bourdon tube through a connection at the back of the case.

The range of the gage is typically from zero p.s.i. to 20 p.s.i., with the scale marked in 2-p.s.i. graduations as shown in figure 12–5.

When installed and connected into an aircraft's de-icing pressure system, the gage reading always remains at zero unless the de-icing system is operating. The gage pointer will fluctuate from zero p.s.i. to approximately 8 p.s.i. under normal conditions, because the de-icer boots are periodically inflated and deflated. This normal fluctuation should not be confused with oscillation.

Diaphragm-Type Pressure Gages

This type of pressure gage uses a diaphragm for measuring pressure. The pressure or suction to be measured is admitted to the pressure-sensitive diaphragm through an opening in the back of the instrument case (figure 12–6).

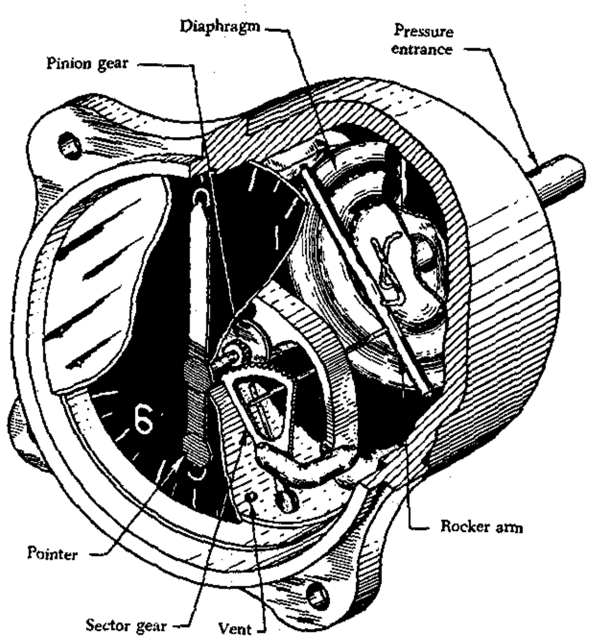


FIGURE 12–6. Diaphragm-type pressure gage.

An opposing pressure, such as that of the atmosphere, is admitted through a vent in the case (figure 12–6). Since the walls of the diaphragm are very thin, an increase of pressure will cause it to expand, and a decrease in pressure will cause it to contract. Any movement of the diaphragm is transferred to the pointer by means of the rocker shaft, sector, and pinion, which are connected to the front side of the diaphragm. This gage is also a differential-pressure measuring device since it indicates the difference between the pressure applied at the vent of the case and the pressure or suction inside the diaphragm.

Suction Gages

Suction gages are used on aircraft to indicate the amount of suction that actuates the air-driven gyroscopic instruments. The spinning rotors of gyroscopic instruments are kept in motion by streams of air directed against the rotor vanes. These airstreams are produced by pumping air out of the instrument cases by the vacuum pump. Atmospheric pressure then forces air into the cases through filters, and it is this air that is directed against the rotor vanes to turn them.

The suction gage indicates whether the vacuum system is working properly. The suction gage case is vented to the atmosphere or to the line of the air filter, and contains a pressure-sensitive diaphragm plus the usual multiplying mechanism which amplifies the movement of the diaphragm and transfers it to the pointer. The reading of a suction gage indicates the difference between atmospheric pressure and the reducd pressure in the vacuum system.

Manifold Pressure Gage

The manifold pressure gage is an important instrument in an aircraft powered by a reciprocating engine. The gage is designed to measure absolute pressure. This pressure is the sum of the air pressure and the added pressure created by the supercharger. The dial of the instrument is calibrated in inches of mercury (Hg).

When the engine is not running, the manifold pressure gage records the existing atmospheric pressure. When the engine is running, the reading obtained on the manifold pressure gage depends on the engine's r.p.m. The manifold pressure gage indicates the manifold pressure immediately before the cylinder intake ports.

The schematic of one type of manifold pressure gage is shown in figure 12–7. The outer shell of the gage protects and contains the mechanism. An opening at the back of the case provides for the connection to the manifold of the engine.

The gage contains an aneroid diaphragm and a linkage for transmitting the motion of the diaphragm to the pointer. The linkage is completely external to the pressure chamber, and thus is not exposed to the corrosive vapors of the manifold. The pressure existing in the manifold enters the sealed chamber through a damping tube, which is a short length of capillary tubing at the rear of the

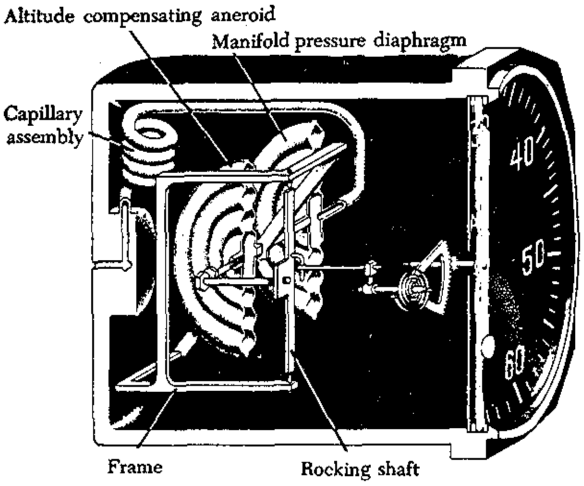


FIGURE 12–7. Manifold pressure gage.

case. This damping tube acts as a safety valve to prevent damage to the instrument by engine backfire. The sudden surge of pressure caused by backfire is considerably reduced by the restricted capillary tubing.

When installing a manifold pressure gage, care should be taken to ensure that the pointer is vertical when registering 30 in. Hg.

When an engine is not running, the manifold pressure gage reading should be the same as the local barometric pressure. It can be checked against a barometer known to be in proper operating condition. In most cases the altimeter in the aircraft can be used, since it is a barometric instrument. With the aircraft on the ground, the altimeter hands should be set to zero and the instrument panel tapped lightly a few times to remove any possible frictional errors. The barometer scale on the altimeter face will indicate local atmospheric pressure when the altimeter hands are at zero. The manifold pressure gage should agree with this pressure reading. If it does not, the gage should be replaced with a gage that is operating properly.

If the pointer fails to respond entirely, the mechanism is, in all probability, defective. The gage should be removed and replaced. If the pointer responds but indicates incorrectly, there may be moisture in the system, obstruction in the lines, a leak in the system, or a defective mechanism.

When doubt exists about which of these items is the cause of the malfunction, the engine should be operated at idle speed and the drain valve (usually located near the gage) opened for a few minutes. This will usually clear the system of moisture. To clear an obstruction, the lines may be disconnected and blown clear with compressed air. The gage mechanism may be checked for leaks by disconnecting the line at the engine end and applying air pressure until the gage indicates 50 in. Hg. Then the line should be quickly closed. A leak is present if the gage pointer returns to atmospheric pressure. If a leak is evident but cannot be found, the gage should be replaced.

PITOT-STATIC SYSTEM

Three of the most important flight instruments are connected into a pitot-static system. These instruments are the airspeed indicator, the altimeter, and the rate-of-climb indicator. Figure 12–8 shows these three instruments connected to a pitot-static tube head.

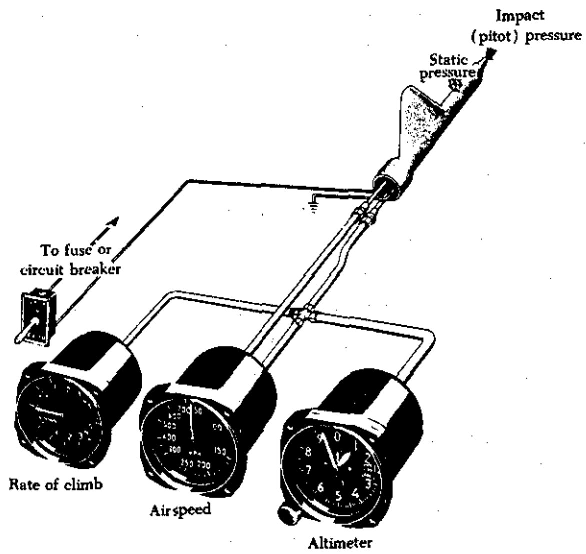


FIGURE 12–8. Pitot-static system.

The pitot-static system head, or pitot-static tube as it is sometimes called, consists of two sections. As shown in figure 12–9, the forward section is open at the front end to receive the full force of the impact air pressure. At the back of this section is a baffle plate to protect the pitot tube from moisture and dirt that might otherwise be blown into it. Moisture can escape through a small drain hole at the bottom of the forward section.

The pitot, or pressure, tube leads back to a chamber in the "shark-fin" projection near the rear of the assembly. A riser, or upright tube, leads the air from this chamber through tubing to the airspeed indicator.

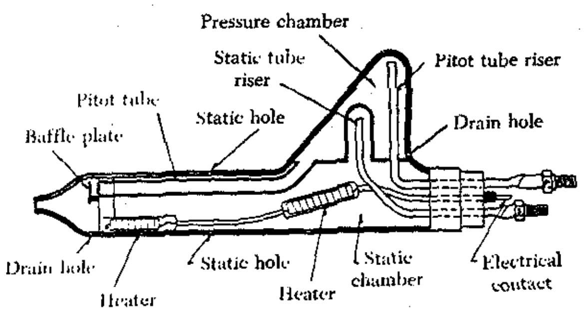


FIGURE 12–9. Pitot-static system head.

The rear, or static, section of the pitot-static tube head is pierced by small openings on the top and bottom surfaces. These openings are designed and located so that this part of the system will provide accurate measurements of atmospheric pressure in a static, or still, condition. The static section contains a riser tube which is connected to the airspeed indicator, the altimeter, and the rate-of-climb indicator.

Many pitot-static tubes are provided with heating elements to prevent icing during flight (figure 12–9). During ice-forming conditions, the electrical heating elements can be turned on by means of a switch in the cockpit. The electrical circuit for the heater element may be connected through the ignition switch. Thus, in case the heater switch is inadvertently left in the "on" position, there will be no drain on the battery when the engine is not operating.

The pitot-static tube head is mounted on the outside of the aircraft at a point where the air is least likely to be turbulent. It is pointed in a forward direction parallel to the aircraft's line of flight. One general type of tube head is designed for mounting on a streamlined mast extending below the nose of the aircraft fuselage. Another type is designed for installation on a boom extending forward of the leading edge of the wing. Both types are shown in figure 12–10. Although there is a slight difference in their construction, they operate identically.

Most pitot-static tubes are manufactured with a union connection in both lines from the head, near the point at which the tube head is attached to the mounting boom or mast (figure 12–10). These connections simplify removal and replacement, and are usually reached through an inspection door in the wing or fuselage. When a pitot-static tube head is to be removed, these connections should be disconnected before any mounting screws and lockwashers are removed.

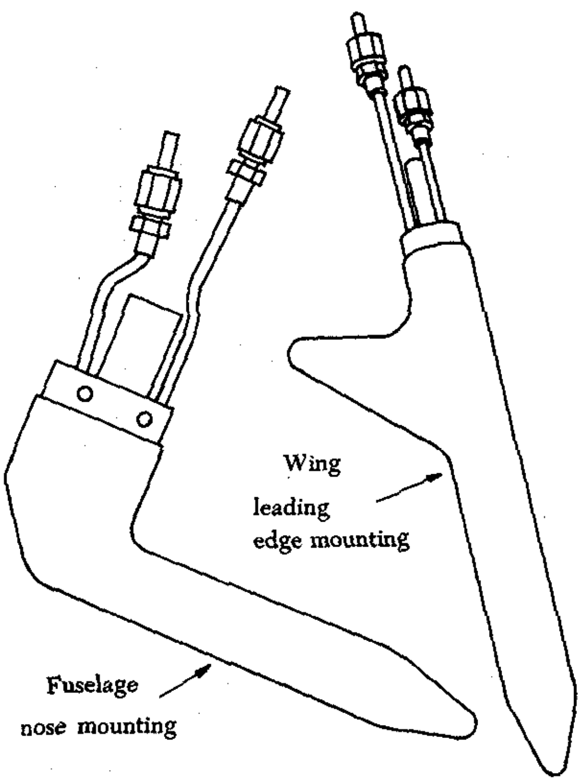


FIGURE 12–10. Pitot-static tube heads.

In many aircraft equipped with a pitot-static tube, an alternate source of static pressure is provided for emergency use. A schematic diagram of a typical system is shown in figure 12–11. As shown in the diagram, the alternate source of static pressure may be vented to the interior of the aircraft.

Another type of pitot-static system provides for the location of the pitot and static sources at sepa-

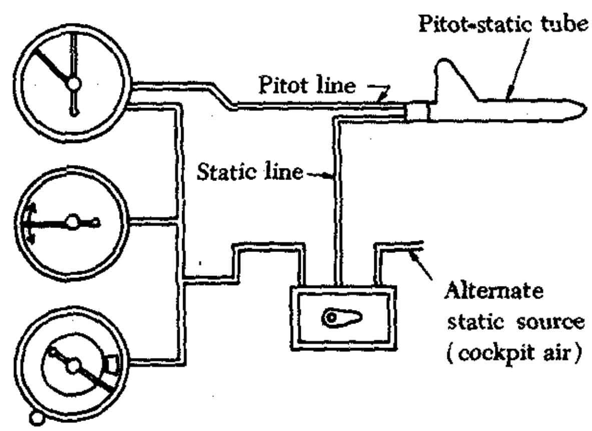


FIGURE 12–11. Pitot-static system with alternate source of static pressure.

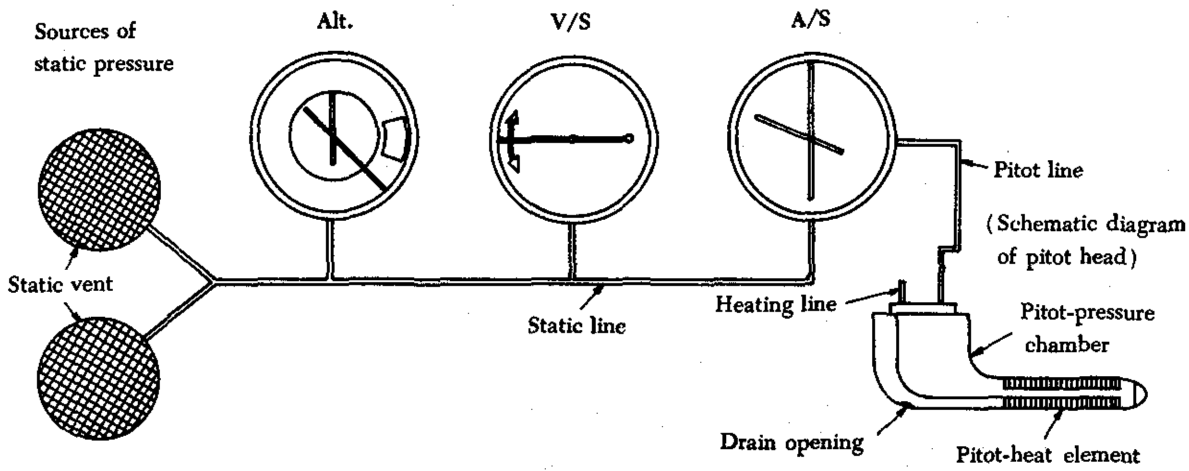


FIGURE 12–12. Pitot-static system with separate sources of pressure.

rate positions on the aircraft. This type of system is illustrated in figure 12–12.

Impact pressure is taken from the pitot-head (figure 12–12) which is mounted parallel to the longitudinal axis of the aircraft and generally in line with the relative wind. The leading edge of the wing, nose section, or vertical stabilizer are the usual mounting positions, since at those points there is usually a minimum disturbance of air due to motion of the aircraft.

Static pressure in this type of pitot-static system is taken from the static line attached to a vent or vents mounted flush with the fuselage or nose section. On aircraft using a flush-mounted static source, there may be two vents, one on each side of the aircraft. This compensates for any possible variation in static pressure on the vents due to erratic changes in aircraft attitude. The two vents are usually connected by a Y-type fitting. In this type of system, clogging of the pitot opening by ice or dirt (or failure to remove the pitot cover) affects the airspeed indicator only.

A pitot-static system used on a pressurized, multi-engine aircraft is shown in figure 12–13. Three additional units, the cabin pressure controller, the cabin differential pressure gage, and the autopilot system are integrated into the static system. Both heated and unheated flushmounted static ports are used.

Altimeters

There are many kinds of altimeters in general use today. However, they are all constructed on the same basic principle as an aneroid barometer. They all have pressure-responsive elements (aneroids) which expand or contract with the pressure change of different flight levels. The heart of an altimeter is its aneroid mechanism (figure 12–14). The expansion or contraction of the aneroid with pressure changes actuates the linkage, and the indicating hands show altitude. Around the aneroid mechanism of most altimeters is a device called the bimetal yoke. As the name implies, this device is composed of two metals and performs the function of compensating for the effect that temperature has on the metals of the aneroid mechanism.

The presentation of altitude by altimeters in current use varies from the multi-pointer type to the drum and single pointer, and the digital counter and single pointer types.

The dial face of the typical altimeter is graduated with numerals from zero to 9 inclusive, as shown in figure 12–15. Movement of the aneroid element is transmitted through a gear train to the three hands on the instrument face. These hands sweep the calibrated dial to indicate the altitude of the aircraft. The shortest hand indicates altitude in tens of thousands of feet; the intermediate hand, in thousands of feet; and the longest hand, in hundreds of feet in 20-ft. increments. A barometric scale, located at the right of the instrument face, can be set by a knob located at the lower left of the instrument case. The barometric scale indicates barometric pressure in inches of mercury.

Since atmospheric pressure continually changes, the barometric scale must be re-set to the local station altimeter setting before the altimeter will indicate the correct altitude of the aircraft above

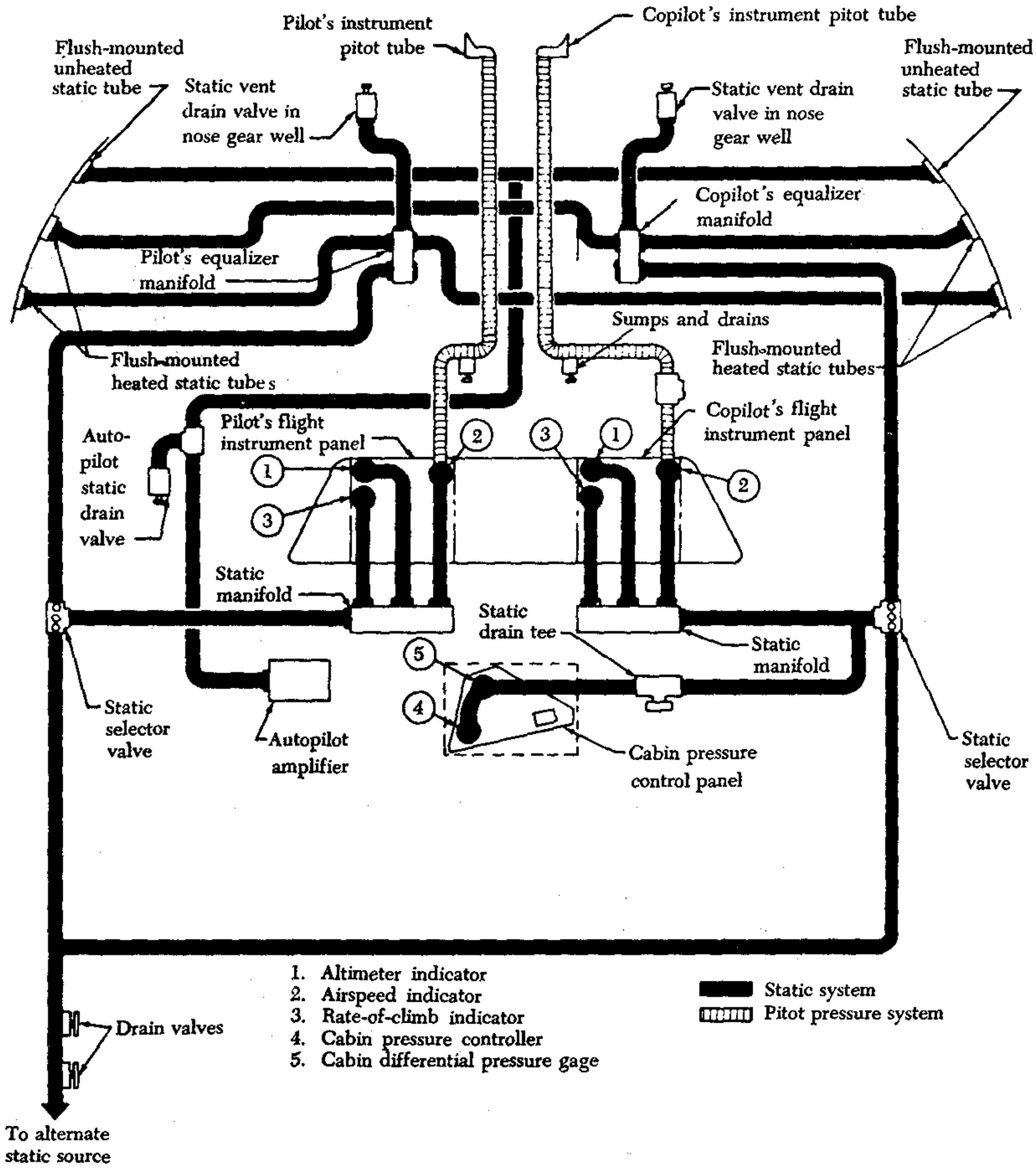


FIGURE 12–13. Schematic of typical pitot-static system on pressurized multi-engine aircraft.

sea level. When the setting knob is turned, the barometric scale, the hands, and the aneroid element move to align the instrument mechanism with the new altimeter setting.

Two setting marks, inner and outer, indicate barometric pressure in feet of altitude. They operate in conjunction with the barometric scale, and indications are read on the altimeter dial. The outer mark indicates hundreds of feet, and the inner mark thousands of feet. Since there is a limit to the graduations which can be placed on the barometric scale, the setting marks are used when the barometric pressure to be read is outside the limits of the scale.

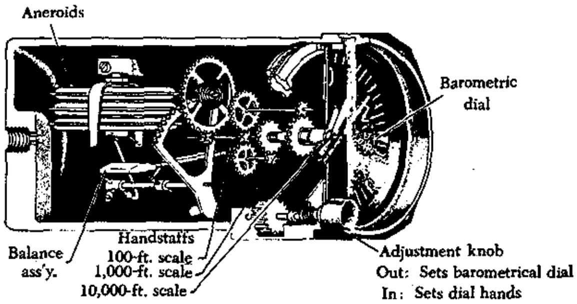


FIGURE 12–14. Mechanism of a sensitive altimeter.

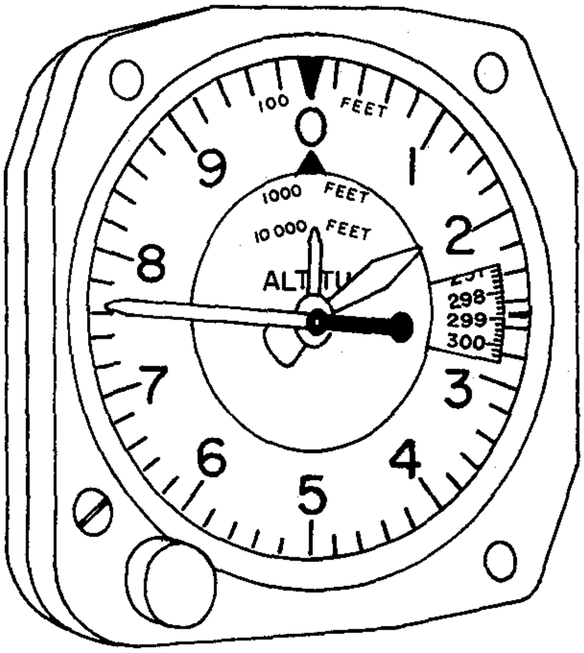


FIGURE 12–15. Sensitive altimeter.

Altimeter Errors

Altimeters are subject to various mechanical errors. A common one is that the scale is not correctly oriented to standard pressure conditions. Altimeters should be checked periodically for scale errors in altitude chambers where standard conditions exist.

Another mechanical error is the hysteresis error. This error is induced by the aircraft maintaining a given altitude for an extended period of time, then suddenly making a large altitude change. The resulting lag or drift in the altimeter is caused by the elastic properties of the materials which comprise the instrument. This error will eliminate itself with slow climbs and descents or after maintaining a new altitude for a reasonable period of time.

In addition to the errors in the altimeter mechanism, another error called installation error affects the accuracy of indications. The error is caused by the change of alignment of the static pressure port with the relative wind. The change of alignment is caused by changes in the speed of the aircraft and in the angle of attack, or by the location of the static port in a disturbed pressure field. Improper installation or damage to the pitot-static tube will also result in improper indications of altitude.

Rate-of-Climb Indicators

The rate-of-climb, or vertical velocity, indicator (figure 12–16) is a sensitive differential pressure gage that indicates the rate at which an aircraft is climbing or descending. The rate-of-climb indicator is connected to the static system and senses the rate of change of static pressure.

The rate of altitude change, as shown on the indicator dial, is positive in a climb and negative

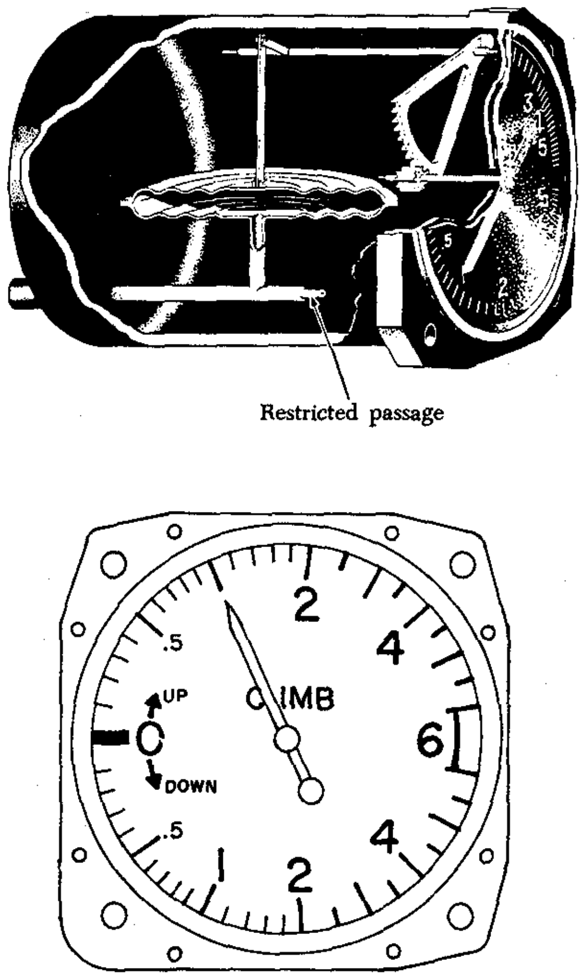


FIGURE 12–16. Typical rate-of-climb indicator.

when descending in altitude. The dial pointer moves in either direction from the zero point, depending on whether the aircraft is going up or down. In level flight the pointer remains at zero.

The operation of a climb indicator is illustrated in figure 12–16. The case of the instrument is airtight except for a small connection through a restricted passage to the static line of the pitot-static system.

Inside the sealed case of the rate-of-climb indicator is a diaphragm with connecting linkage and gearing to the indicator pointer. Both the diaphragm and the case receive air at atmospheric pressure from the static line. When the aircraft is on the ground or in level flight, the pressures inside the diaphragm and the instrument case remain the same and the pointer is at the zero indication. When the aircraft climbs the pressure inside the diaphragm decreases but, due to the metering action of the restricted passage, the case pressure will remain higher and cause the diaphragm to contract. The diaphragm movement actuates the mechanism, causing the pointer to indicate a rate of climb.

When the aircraft levels off, the pressure in the instrument case is equalized with the pressure in the diaphragm. The diaphragm returns to its neutral position and the pointer returns to zero.

In a decent, the pressure conditions are reversed. The diaphragm pressure immediately becomes greater than the pressure in the instrument case. The diaphragm expands and operates the pointer mechanism to indicate the rate of descent.

When the aircraft is climbing or descending at a constant rate, a definite ratio between the diaphragm pressure and the case pressure is maintained through the calibrated restricted passage, which requires approximately 6 to 9 sec. to equalize the two pressures, causing a lag in the proper reading. Any sudden or abrupt changes in the aircraft's attitude may cause erroneous indications due to the sudden change of airflow over the static ports.

The instantaneous rate-of-climb indicator is a more recent development which incorporates acceleration pumps to eliminate the limitations associated with the calibrated leak. For example, during an abrupt climb, vertical acceleration causes the pumps to supply extra air into the diaphragm to stabilize the pressure differential without the usual lag time. During level flight and steady-rate climbs and descents, the instrument operates on the same principles as the conventional rate-of-climb indicator.

A zero-setting system, controlled by a setscrew or an adjusting knob permits adjustment of the pointer to zero. The pointer of an indicator should indicate zero when the aircraft is on the ground or maintaining a constant pressure level in flight.

Airspeed Indicator

Airspeed indicators are sensitive pressure gages which measure the difference between the pitot and static pressures, and present such difference in terms of indicated airspeed. Airspeed indicators are made by various manufacturers and vary in their mechanical construction. However, the basic construction and operating principle is the same for all types.

The airspeed indicator (figure 12–17) is a sensitive, differential pressure gage which measures and indicates promptly the differential between the impact and the static air pressures surrounding an airplane at any moment of flight. The airspeed indicator consists primarily of a sensitive metallic diaphragm whose movements, resulting from the slightest difference in impact and static air pressures, are multiplied by means of a link, a rocking shaft, a sector with hairspring and pinion, and a tapered shaft to impart rotary motion to the pointer, which indicates the aircraft velocity on the dial face in terms of knots or m.p.h.

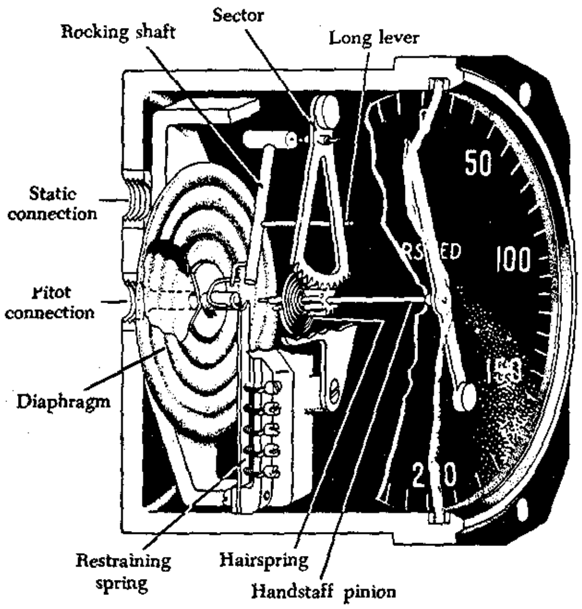


FIGURE 12–17. Airspeed indicator.

Most airspeed indicators are marked to show speed limitations at a glance. The never-exceed velocity is designated by a red radial line. A yellow arc designates the cautionary range, and a white arc is used to indicate the range of permissible limits of flap operation.

The dial numbers used on different airspeed indicators are indicative of the type of aircraft in which they are used; for example, an airspeed indicator with a range of zero to 160 knots is commonly used in many light aircraft. Other types, such as a 430-knot indicator, are used on larger and faster aircraft.

Another type of airspeed indicator in use is the maximum allowable airspeed indicator shown in figure 12–18. This indicator includes a maximum allowable needle, which shows a decrease in maximum allowable airspeed with an increase in altitude. It operates from an extra diaphragm in the airspeed indicator which senses changes in altitude and measures this change on the face of the instrument. Its purpose is to indicate maximum allowable indicated airspeed at any altitude.

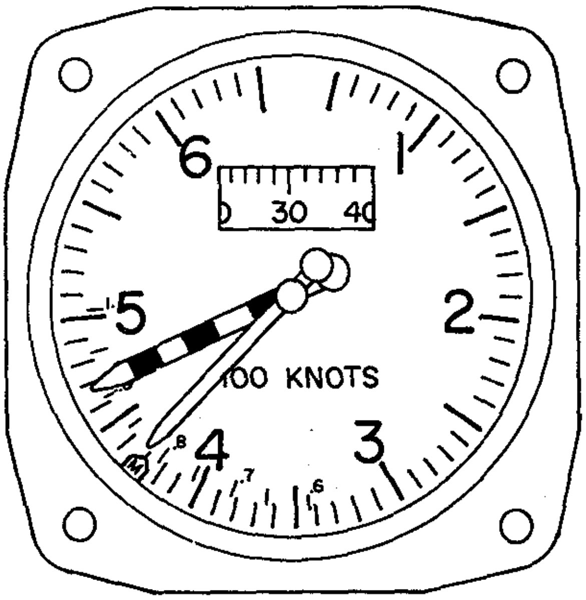


FIGURE 12–18. Maximum allowable airspeed indicator.

The type of airspeed indicator known as a true airspeed indicator is shown in figure 12–19. It uses an aneroid, a differential pressure diaphragm, and a bulb temperature diaphragm, which respond respectively to changes in barometric pressure, impact pressure, and free air temperature. The actions of the diaphragms are mechanically resolved to indicate true airspeed in knots. A typical true airspeed indicator is designed to indicate true airspeed from 1,000 ft. below sea level to 50,000 ft. above sea level under free air temperature conditions from +40° to —60° C.

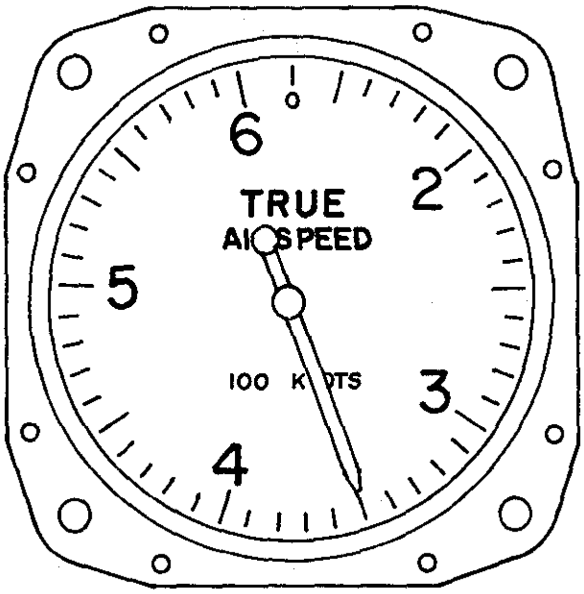


FIGURE 12–19. True airspeed indicator.

Mach Indicator

Machmeters indicate the ratio of aircraft speed to the speed of sound at the particular altitude and temperature existing at any time during flight.

Construction of a Mach indicator is much the same as that of an airspeed indicator. It will usually contain a differential pressure diaphragm which senses pitot-static pressure, and an aneroid diaphragm which senses static pressure. By mechanical means, changes in pressures are then displayed on the instrument face in terms of Mach numbers.

The Machmeter shown in figure 12–20A is designed to operate in the range of 0.3 to 1.0 Mach and at altitudes from zero to 50,000 feet. The Machmeter shown in figure 12–20B is designed to operate in the range of 0.5 to 1.5 at altitudes up to 50,000 feet.

Combined Airspeed/Mach Indicator

Combined airspeed/Mach indicators are provided for aircraft where instrument space is at a premium and it is desirable to present information on a combined indicator. These instruments show indicated

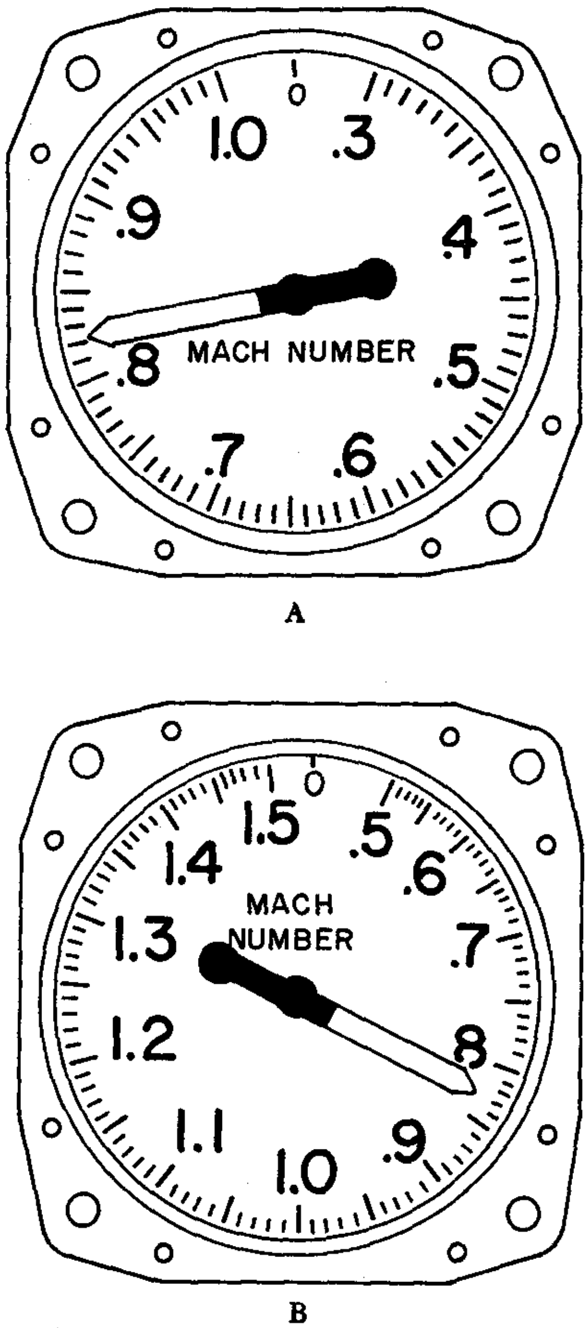


FIGURE 12–20. Machmeters.

airspeed, Mach, and limiting Mach by use of impact and static pressures and an altitude aneroid.

These combined units utilize a dual-pointed needle which shows airspeed on a fixed scale and Mach indication on a rotating scale. A knurled knob located on the lower portion of the instrument is provided to set a movable index marker to reference a desired speed. A combined airspeed/Mach indicator is shown in figure 12–21.

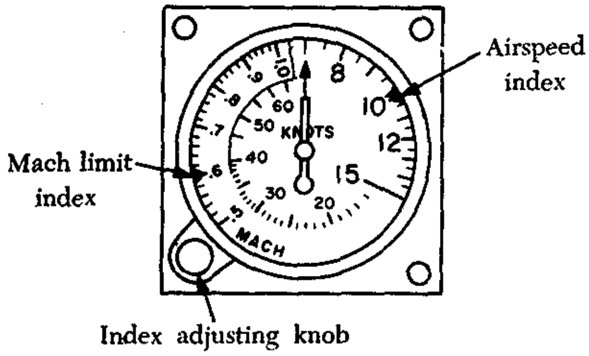


FIGURE 12–21. Combined airspeed/Mach indicator.

MAINTENANCE OF PITOT-STATIC SYSTEMS

The specific maintenance instructions for any pitot-static system are usually detailed in the applicable aircraft manufacturer's maintenance manual. However, there are certain inspections, procedures, and precautions to be observed that apply to all systems.

Pitot tubes and their supporting masts should be inspected for security of mounting and evidence of damage. Checks should also be made to ensure that electrical connections are secure. The pitot pressure entry hole, drain holes, and static holes or ports should be inspected to ensure that they are unobstructed. The size of the drain holes and static holes is aerodynamically critical. They must never be cleared of obstruction with tools likely to cause enlargement or burring.

Heating elements should be checked for functioning by ensuring that the pitot tube begins to warm up when the heater is switched "on." If an ammeter or loadmeter is installed in the circuit, a current reading should be taken.

The inspections to be carried out on the individual instruments are primarily concerned with security, visual defects, and proper functioning. The zero setting of pointers must also be checked. At the time of inspecting the altimeter, the barometric pressure scale should be set to read field barometric pressure. With this pressure set, the instrument should read zero within the tolerances specified for the type installed. No adjustment of any kind can be made, if the reading is not within limits, the instrument must be replaced.

Leak Testing Pitot-Static Systems

Aircraft pitot-static systems must be tested for leaks after the installation of any component parts, when system malfunction is suspected, and at the periods specified in the Federal Aviation Regulations.

The method of leak testing and the type of equipment to use depends on the type of aircraft and its pitot-static system. In all cases, pressure and suction must be applied and released slowly to avoid damage to the instruments. The method of testing consists basically of applying pressure and suction to pressure heads and static vents respectively, using a leak tester and coupling adapters. The rate of leakage should be within the permissible tolerances prescribed for the system. Leak tests also provide a means of checking that the instruments connected to a system are functioning properly. However, a leak test does not serve as a calibration test.

Upon completion of the leak test, be sure that the system is returned to the normal flight configuration. If it was necessary to blank off various portions of a system, check to be sure that all blanking plugs, adapters, or pieces of adhesive tape have been removed.

TURN-AND-BANK INDICATOR

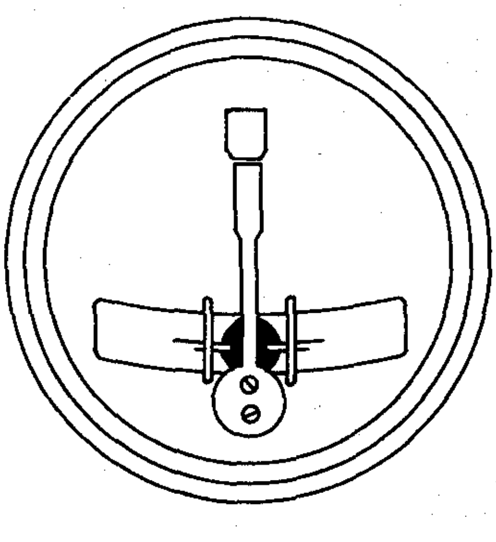
The turn-and-bank indicator, figure 12–22, also referred to as the turn-and-slip or needle-and-ball indicator, shows the correct execution of a bank and turn and indicates the lateral attitude of the aircraft in level flight.

The turn needle is operated by a gyro, driven either by a vacuum, air pressure, or electricity. The turn needle indicates the rate, in number of degrees per second, at which an aircraft is turning about its vertical axis. It also provides information on the amount of bank. The gyro axis is horizontally mounted so that the gyro rotates up and away from the pilot. The gimbal around the gyro is pivoted fore and aft.

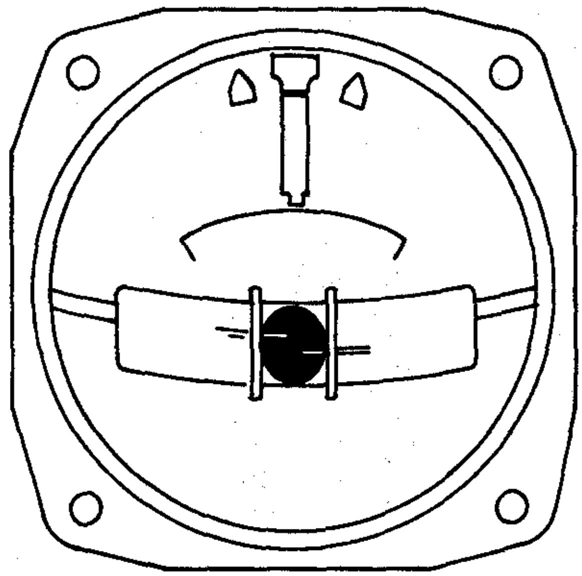
Gyroscopic precession causes the rotor to tilt when the aircraft is turned. Due to the direction of rotation, the gyro assembly tilts in the opposite direction from which the aircraft is turning. This prevents the rotor axis from becoming vertical to the earth's surface. The linkage between the gyro assembly and the turn needle, called the reversing mechanism, causes the needle to indicate the proper direction of turn.

Power for the electric gyro may be supplied from either an a.c. or d.c. source.

The principal value of the electric gyro in light aircraft is its safety factor. In single-engine aircraft equipped with vacuum-driven attitude and heading indicators, the turn needle is commonly operated by an electric gyro. In the event of vacuum system failure and loss of two gyro instruments, the pilot still has a reliable standby instrument for emergency operation. Operated on current directly from the battery, the electric turn indicator is reliable as long as current is available, regardless of generator or vacuum system malfunction. In the electric instrument, the gyro is a small electric motor and flywheel. Otherwise both electric and vacuum-driven turn-needles are de-



Two minute turn indicator



Four minute turn indicator

FIGURE 12–22. Two types of turn-and-bank indicators.

signed to use the same gyroscopic principle of precession.

Power for the suction-driven turn needle is regulated by a restrictor installed between the main suction line and the instrument to produce a desired suction and rotor speed. Since the needle measures the force of precession, excessivly high or low vacuum results in unreliable turn-needle operation. For a specific rate of turn, low vacuum produces less than normal rotor speed and, therefore, less needle deflection for this specific rate of turn. The reverse is true for the condition of high vacuum.

Of the two types of turn needles shown in figure 12–22, the 2-min. turn indicator is the older. If the instrument is accurately calibrated; a single needle-width deflection on the 2-min. indicator means that the aircraft is turning at 3° per sec., or standard (2 min. for a 360° turn). On the 4-min. indicator, a single needle-width deflection shows when the aircraft is turning at 1½° per sec., or half standard rate (4 min. for a 360° turn). The 4-min. turn indicator was developed especially for high-speed aircraft.

The slip indicator (ball) part of the instrument is a simple inclinometer consisting of a sealed, curved glass tube containing kerosene and a black agate or a common steel ball bearing, which is free to move inside the tube. The fluid provides a damping action, ensuring smooth and easy movement of the ball. The tube is curved so that in a horizontal position the ball tends to seek the lowest point. A small projection on the left end of the tube contains a bubble of air which compensates for expansion of the fluid during changes in temperature. Two strands of wire wound around the glass tube fasten the tube to the instrument case and also serve as reference markers to indicate the correct position of the ball in the tube. During coordinated straight-and-level flight, the force of gravity causes the ball to rest in the lowest part of the tube, centered between the reference wires.

Maintenance Practices for Turn-and-Bank Indicators

Errors in turn needle indications are usually due to insufficient or excessive rotor speed or inaccurate adjustment of the calibrating spring. There is no practical operational test or checkout of this instrument, other than visually noting that the indicator pointer and the ball are centered.

SYNCHRO-TYPE REMOTE INDICATING INSTRUMENTS

A synchro system is an electrical system used for transmitting information from one point to another. Most position-indicating instruments are designed around a synchro system. The word "synchro" is a shortened form of synchronous, and refers to any one of a number of electrical devices capable of measuring and indicating angular deflection. Synchro systems are used as remote position indicators for landing gear and flap systems, in autopilot systems, in radar systems, and many other remote-indicating applications. There are different types of synchro systems. The three most common are: (1) Autosyn, (2) selsyn, and (3) magnesyn. These systems are similar in construction and all operate on identical electrical and mechanical principles.

D.C. Selsyn Systems

The d.c. selsyn system is a widely used electrical method of indicating a remote mechanical condition. Specifically, d.c. selsyn systems can be used to show the movement and position of retractable landing gear, wing flaps, cowl flaps, oil cooler doors, or similar movable parts of the aircraft.

A selsyn system consists of a transmitter, an indi-

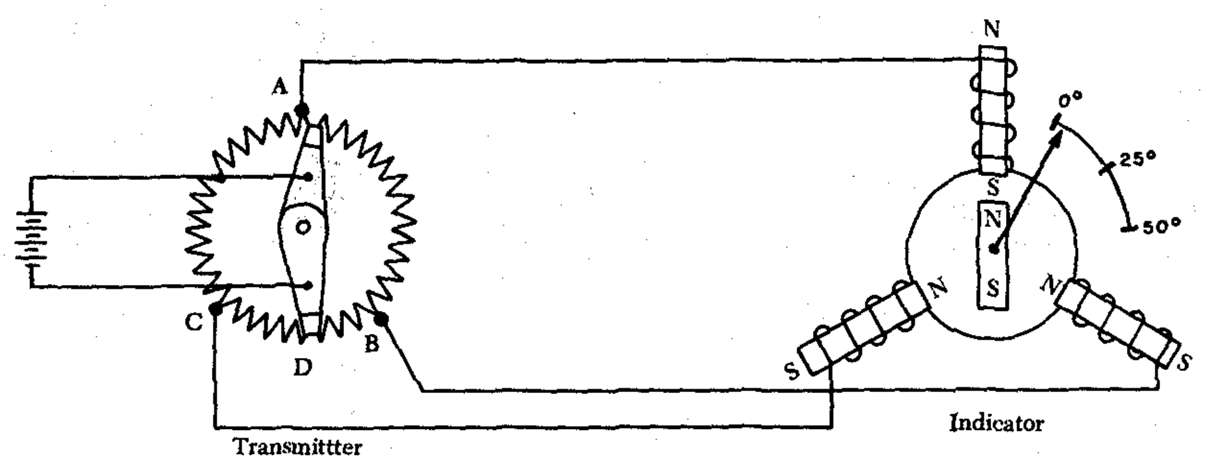


FIGURE 12–23. Schematic diagram of a d.c. selsyn system.

cator and connecting wires. The voltage required to operate the selsyn system is supplied from the aircraft's electrical system.

A selsyn system is shown schematically in figure 12–23. The transmitter consists of a circular resistance winding and a rotatable contact arm. The rotatable contact arm turns on a shaft in the center of the resistance winding. The two ends of the arm, or brushes, always touch the winding on opposite sides. The shaft to which the contact arm is fastened protrudes through the end of the transmitter housing and is attached to the unit (flaps, landing gear, etc.) whose position is to be transmitted. The transmitter is usually connected to the unit through a mechanical linkage. As the unit moves, it causes the transmitter shaft to turn. Thus, the arm can be turned so that voltage can be applied at any two points around the circumference of the winding.

As the voltage at the transmitter taps is varied, the distribution of currents in the indicator coils varies and the direction of the resultant magnetic field across the indicator is changed. The magnetic field across the indicating element corresponds in position to the moving arm in the transmitter. Whenever the magnetic field changes direction, the polarized motor turns and aligns itself with the new position of the field. The rotor thus indicates the position of the transmitter arm.

When the d.c. selsyn system is used to indicate the position of landing gear, an additional circuit is connected to the transmitter winding, which acts as a lock-switch circuit. The purpose of this circuit is to show when the landing gear is up and locked, or down and locked. Lock switches are shown connected into a three-wire system in figure 12–24.

A resistor is connected between one of the taps of the transmitter at one end and to the individual lock switches at the other end. When either lock switch is closed, the resistance is added into the transmitter circuit to cause an unbalance in one section of the transmitter winding. This unbalance causes the current flowing through one of the indicator coils to change. The resultant movement in the indicator pointer shows that the lock switch has been closed. The lock switch is mechanically connected to the landing gear up- or down-locks, and when the landing gear locks either up or down, it closes the lock switch connected to the selsyn transmitter. This locking of the landing gear is repeated on the indicator.

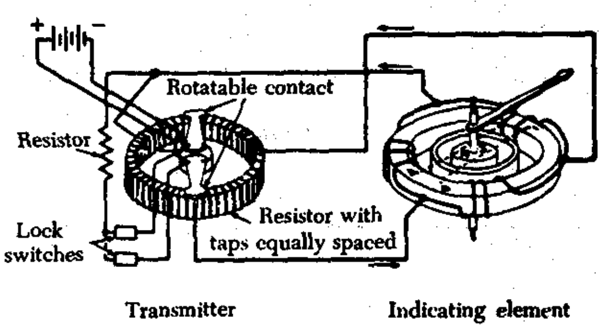


FIGURE 12–24. A double-lock switch in a three-wire selsyn system.

Magnesyn System

The magnesyn system is an electrical self-synchronous device used to transmit the direction of a magnetic field from one coil to another. The magnesyn position system is essentially a method of measuring the extent of the movement of such elements as the wing and cowl flaps, trim tabs, landing gear, or other control surfaces. The two main units of the system are the transmitter and the indicator (figure 12–25).

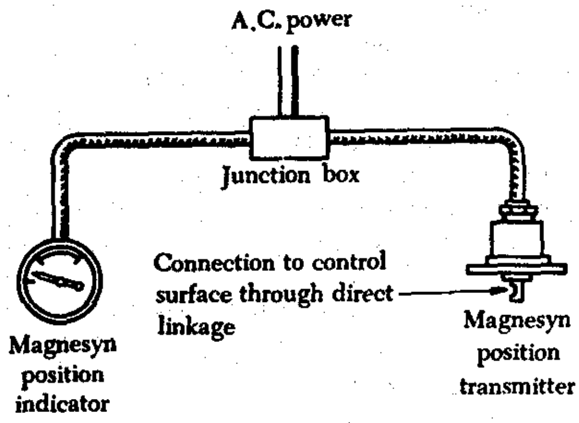


FIGURE 12–25. Magnesyn position-indicating system.

In a magnesyn transmitter a soft iron ring is placed around a permanent magnet so that most of the magnet's lines of force pass within the ring. This circular core of magnetic material is provided with a single continuous electrical winding of fine wire. Figure 12–26 shows an electrical wiring schematic of a magnesyn system. The circular core of magnetic material and the winding are the essential components of the magnesyn stator. The rotor consists of the permanent magnet.

The movement of the control surface of the aircraft causes a proportional movement of the transmitter shaft. This in turn causes a rotary displacement of the magnet. Varying voltages are set up in

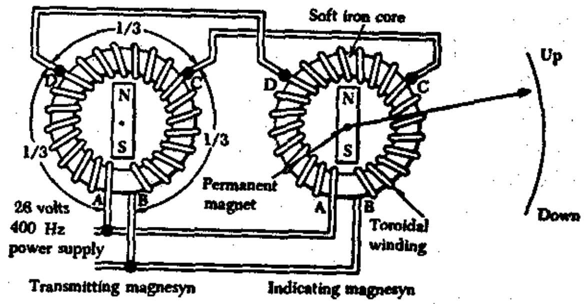


FIGURE 12-26. Magnesyn system.

the magnesyn stator, depending on the position of the magnet. The voltage is transmitted to a magnesyn indicator which indicates on a dial the values received from the transmitter. The indicator consists essentially of a magnesyn, a graduated dial, and a pointer. The pointer is attached to the shaft and the shaft is attached to the magnet; thus, movement of the magnet causes movement of the pointer.

REMOTE-INDICATING FUEL AND OIL PRESSURE GAGES

Fuel and oil pressure indications can be conveniently obtained through use of the various synchro systems. The type of synchro system used may be the same for either fuel or oil pressure measurement; however, an oil system transmitter is usually not interchangeable with a fuel system transmitter.

A typical oil pressure indicating system is shown in figure 12–27. A change in oil pressure introduced into the synchro transmitter causes an electrical signal to be transmitted through the interconnecting wiring to the synchro receiver. This signal causes the receiver rotor and the indicator pointer to move a distance proportional to the amount of pressure exerted by the oil.

Most oil pressure transmitters are composed of two main parts, a bellows mechanism for measuring pressure and a synchro assembly. The pressure of the oil causes linear displacement of the synchro rotor. The amount of displacement is proportional to the pressure, and varying voltages are set up in the synchro stator. These voltages are transmitted to the synchro indicator.

In some installations, dual indicators are used to obtain indications from two sources. On some aircraft, both oil and fuel pressure transmitters are joined through a junction and operate a synchro oil and fuel pressure indicator (dual side-by-side), thus combining both gages in one case.

CAPACITOR-TYPE FUEL QUANTITY SYSTEM

The capacitor-type fuel quantity system is an electronic fuel measuring device that accurately determines the weight of the fuel in the tanks of an aircraft. The basic components of the system are an indicator, a tank probe, a bridge unit, and an amplifier. In some systems the bridge unit and amplifier are one unit, mounted in the same box. More recent systems have been designed with the bridge unit and a transistorized amplifier built into the instrument case.

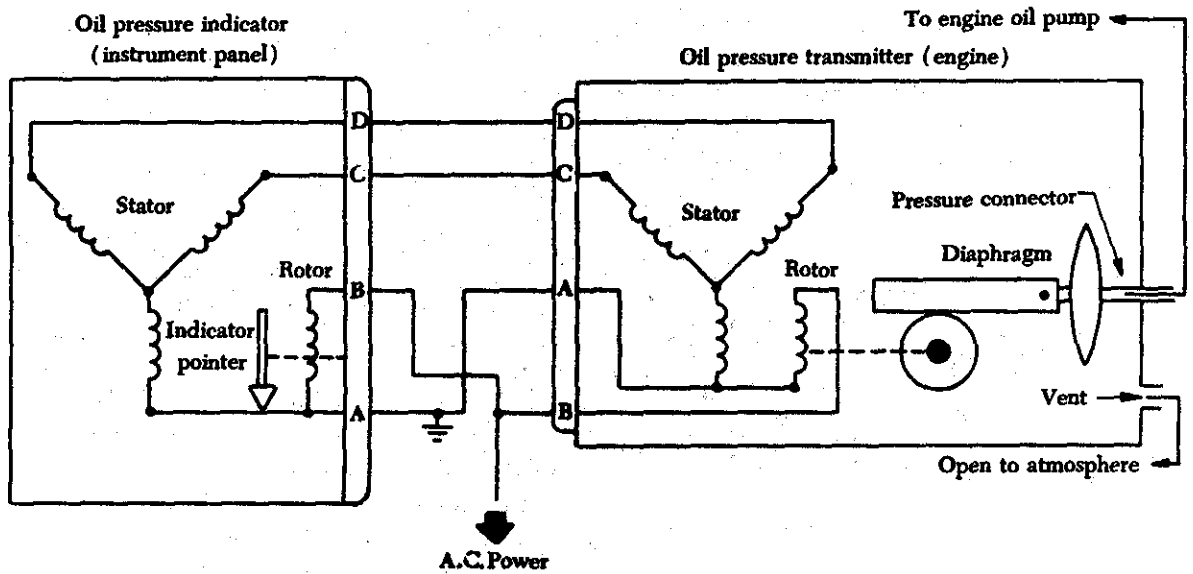


FIGURE 12–27. Oil pressure synchro system.

The fuel quantity indicator shown in figure 12–28 is a sealed, self balancing, motor-driven instrument containing a motor, pointer assembly, transistorized amplifier, bridge circuit, and adjustment potentiometers. A change in the fuel quantity of a tank causes a change in the capacitance of the tank unit. The tank unit is one arm of a capacitance bridge circuit. The voltage signal resulting from the unbalanced bridge is amplified by a phase-sensitive amplifier in the power unit. This signal energizes one winding of a two-phase induction motor in the indicator. The induction motor drives the wiper or a rebalancing potentiometer in the proper direction to balance the bridge, and at the same time positions an indicator pointer to show the quantity of fuel remaining in the tank.

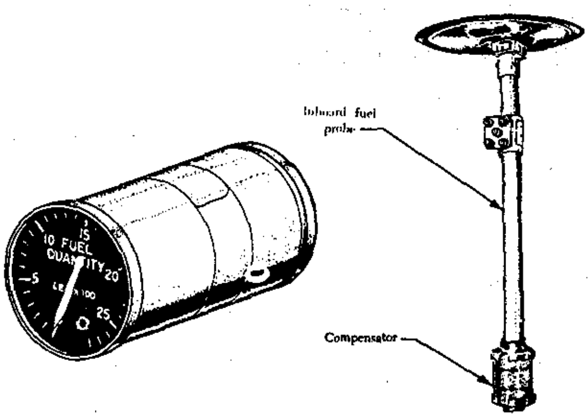


FIGURE 12–28. Indicator and probe of a capacitor type fuel quantity system.

A simplified version of a tank unit is shown in figure 12–29. The capacitance of a capacitor depends on three factors: (1) The area of the plates, (2) the distance between the plates, and (3) the dielectric constant of the material between the plates. The only variable factor in the tank unit is the dielectric of the material between the plates. When the tank is full, the dielectric material is all fuel. Its dielectric constant is about 2.07 at 0° C., compared to a dielectric constant of 1 for air. When the tank is half full, there is air between the upper half of the plates and fuel between the lower half. Thus, the capacitor has less capacitance than it had when the tank was full. When the tank is empty, there is only air between the plates; consequently, the capacitance is still less. Any change in fuel quantity between full and empty will produce a corresponding change in capacitance.

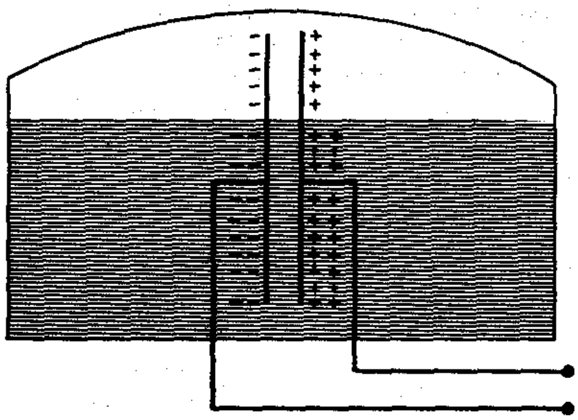


FIGURE 12–29. Simplified capacitance-tank circuit.

A simplified capacitance bridge circuit is shown in figure 12–30. The fuel tank capacitor and a fixed reference capacitor are connected in series across a transformer secondary winding. A voltmeter is connected from the center of the transformer winding to a point between the two capacitors. If the two capacitances are equal, the voltage drop across them will be equal, and the voltage between the center tap and point P will be zero. As the fuel quantity increases, the capacitance of the tank unit increases, causing more current to flow in the tank unit leg of the bridge circuit. This will cause a voltage to exist across the voltmeter that is in phase with the voltage applied to the transformer. If the quantity of fuel in the tank decreases, there will be a smaller flow of current in the tank unit leg of the bridge.

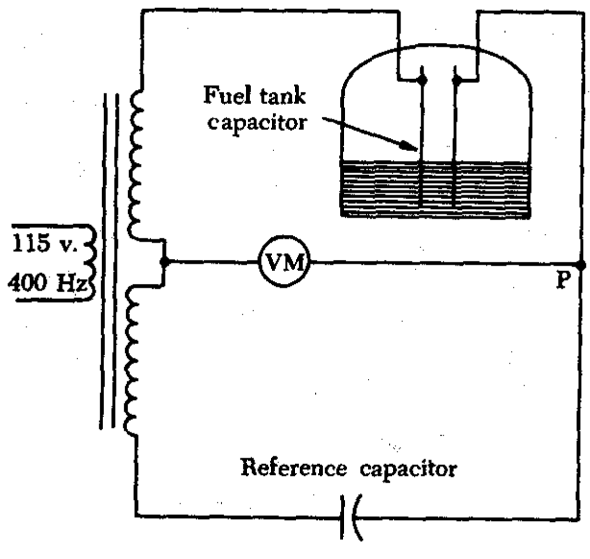


FIGURE 12–30. Simplified capacitance bridge circuit.

The voltage across the voltmeter will now be out of phase with the voltage applied to the transformer.

In an actual capacitor type fuel gage, the input to a two-stage amplifier is connected in place of the voltmeter. It amplifies the signal resulting from an unbalance in the bridge circuit. The output of the amplifier energizes a winding of the two-phase indicator motor. The other motor winding, called the line phase winding, is constantly energized by the same voltage that is applied to the transformer in the bridge circuit, but its phase is shifted 90° by a series capacitor. As a result, the indicator motor is phase sensitive; that is, it will operate in either direction, depending on whether the tank unit capacitance is increasing or decreasing.

As the tank unit capacitance increases or decreases because of a change in fuel quantity, it is necessary to readjust the bridge circuit to a balanced condition so the indicator motor will not continue to change the position of the indicating needle. This is accomplished by a balancing potentiometer connected across one-half of the transformer secondary, as shown in figure 12–31. The indicator motor drives this potentiometer wiper in the direction necessary to maintain continuous balance in the bridge.

The circuit shown in figure 12–31 is a self-balancing bridge circuit. An "empty" calibrating potentiometer and a "full" calibrating potentiometer are connected across portions of the transformer secondary winding at opposite ends of the winding. These potentiometers may be adjusted to balance the bridge voltages over the entire empty-to-full capacitance range of a specific system.

In some installations where the indicator shows the contents of only one tank and where the tank is fairly symmetrical, one unit is sufficient. However, for increased accuracy in peculiarly shaped fuel tanks, two or more units are connected in parallel to minimize the effects of changes in aircraft attitude and sloshing of fuel in the tanks.

ANGLE-OF-ATTACK INDICATOR

The angle-of-attack indicating system detects the local angle of attack of the aircraft from a point on the side of the fuselage and furnishes reference

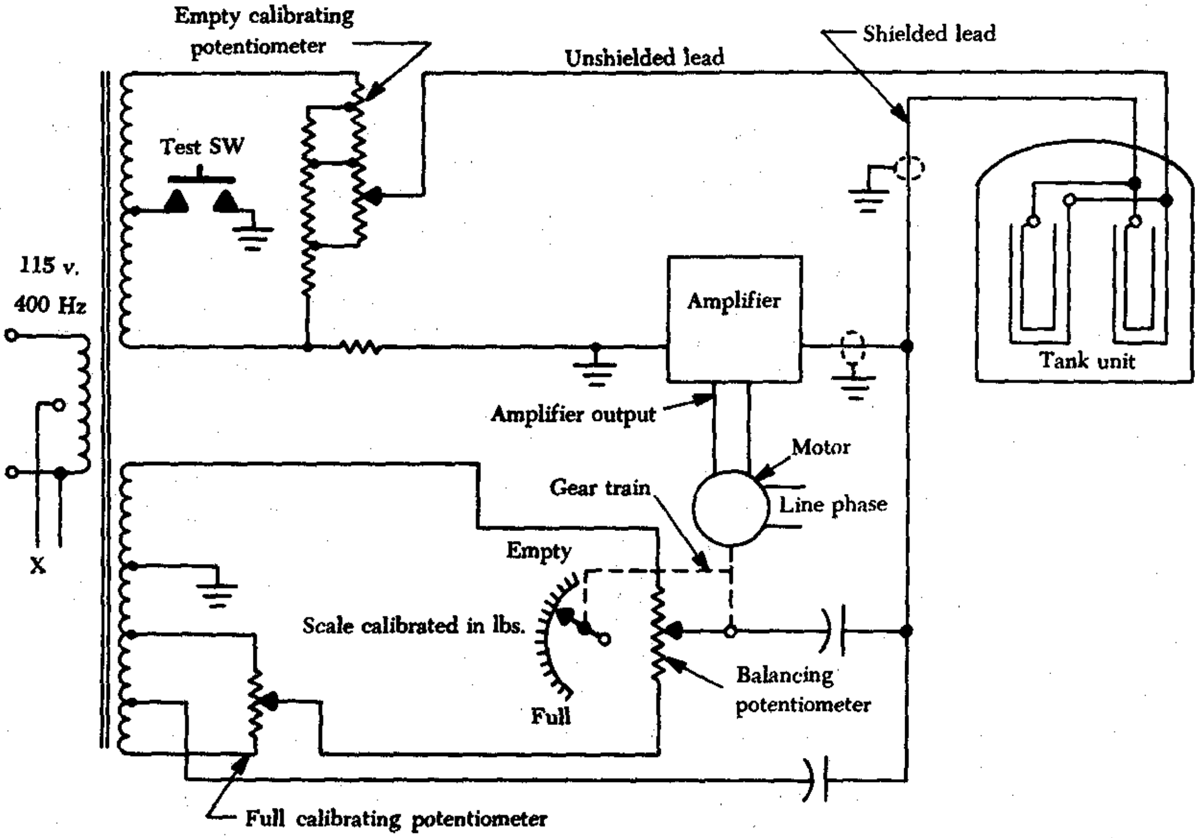


FIGURE 12–31. Self-balancing bridge circuit.

information for the control and actuation of other units and systems in the aircraft. Signals are provided to operate an angle-of-attack indicator (figure 12–32), located on the instrument panel, where a continuous visual indication of the local angel of attack is displayed.

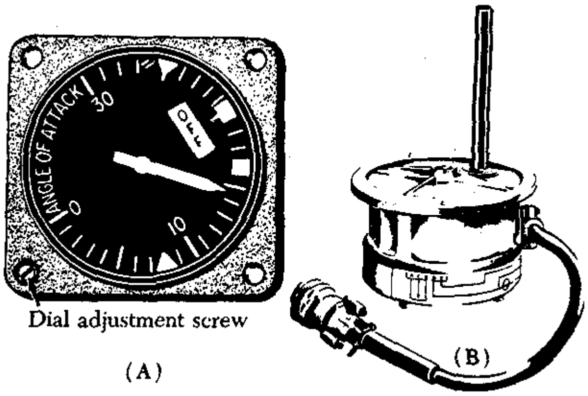


FIGURE 12–32. Angle-of-attack system. (A) Indicator (B) Transmitter.

A typical angle-of-attack system provides electrical signals for the operation of a rudder pedal shaker, which warns the operator of an impending stall when the aircraft is approaching the critical stall angle of attack. Electrical switches are actuated at the angle-of-attack indicator at various preset angles of attack.

The angle-of-attack indicating system consists of an airstream direction detector (transmitter) (figure 12–32B), and an indicator located on the instrument panel. The airstream direction detector contains the sensing element which measures local airflow direction relative to the true angle of attack by determining the angular difference between local airflow and the fuselage reference plane. The sensing element operates in conjunction with a servo-driven balanced bridge circuit which converts probe positions into electrical signals.

The operation of the angle-of-attack indicating system is based on detection of differential pressure at a point where the airstream is flowing in a direction that is not parallel to the true angle of attack of the aircraft. This differential pressure is caused by changes in airflow around the probe unit. The probe extends through the skin of the aircraft into the airstream.

The exposed end of the probe contains two parallel slots which detect the differential airflow pressure (figure 12–33). Air from the slots is transmitted through two separate air passages to separate compartments in a paddle chamber. Any differential pressure, caused by misalignment of the probe with respect to the direction of airflow, will cause the paddles to rotate. The moving paddles will rotate the probe, through mechanical linkage, until the pressure differential is zero. This occurs when the slots are symmetrical with the airstream direction.

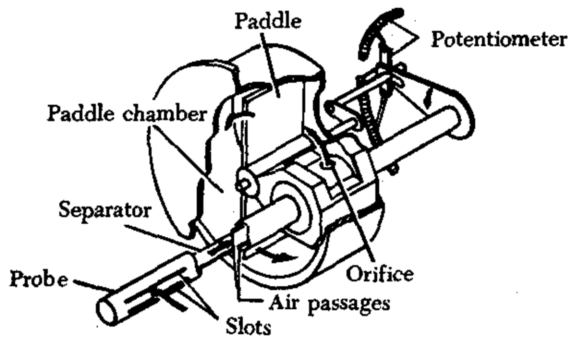


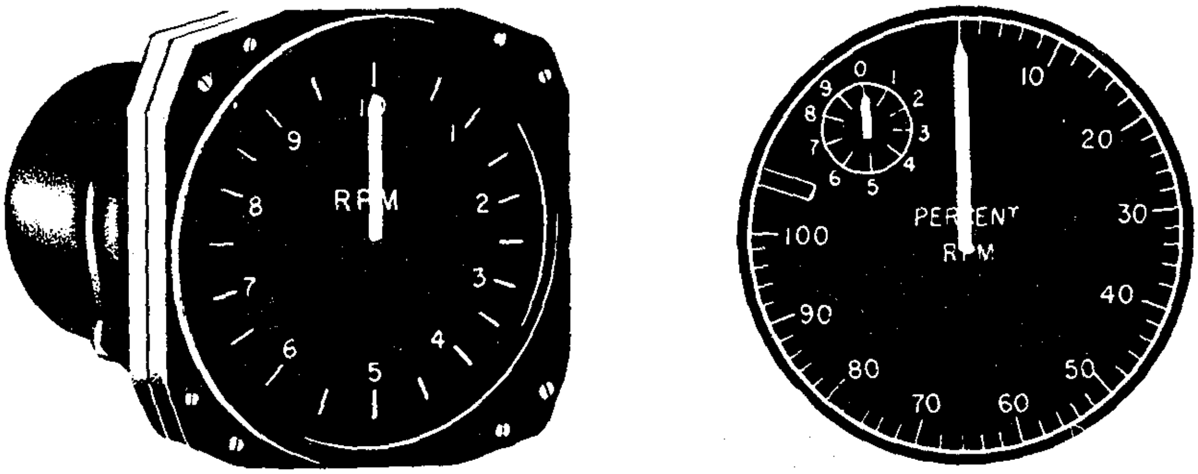
FIGURE 12–33. Airstream direction detector.

Two electrically separate potentiometer wipers, rotating with the probe, provide signals for remote indications. Probe position, or rotation, is converted into an electrical signal by one of the potentiometers which is the transmitter component of a self-balancing bridge circuit. When the angle of attack of the aircraft is changed and, subsequently, the position of the transmitter potentiometer is altered, an error voltage exists between the transmitter potentiometer and the receiver potentiometer the indicator. Current flows through a sensitive polarized relay to rotate a servomotor in the indicator. The servomotor drives a receiver/potentiometer in the direction required to reduce the voltage and restore the circuit to an electrically balanced condition. The indicating pointer is attached to, and moves with, the receiver/potentiometer wiper arm to indicate on the dial the relative angle of attack.

TACHOMETERS

The tachometer indicator is an instrument for indicating the speed of the crankshaft of a reciprocating engine and the speed of the main rotor assembly of a gas turbine engine.

The dials of tachometer indicators used with reciprocating engines are calibrated in r.p.m.; those used with turbine engines are calibrated in percentage of r.p.m. being used, based on the takeoff r.p.m. Figure 12–34 shows a typical dial for each of the indicators just described.



(A) (B)

FIGURE 12–34. Tachometer. (A) Reciprocating engine type. (B) Turbine engine type.

There are two types of tachometer systems in wide use today: (1) The mechanical indicating system, and (2) the electrical indicating system.

Mechanical Indicating Systems

Mechanical indicating systems consist of an indicator connected to the engine by a flexible drive shaft. The indicator contains a flyweight assembly coupled to a gear mechanism that drives a pointer. As the drive shaft rotates, centrifugal force acts on the flyweights assembly and moves them to an angular position. This angular position varies with the r.p.m. of the engine. Movement of the flyweights is transmitted through the gear mechanism to the pointer. The pointer rotates to indicate the r.p.m. of the engine on the tachometer indicator.

Electric Indicating Systems

A number of different types and sizes of tachometer generators and indicators are used in aircraft electrical tachometer systems. Generally, the varous types of tachometer indicators and generators operate on the same basic principle. Thus, the system described will be representative of most electrical tachometer systems; the manufacturer's instructions should always be consulted for details of a specific tachometer system.

The typical tachometer system (figure 12–35) is a three-phase a.c. generator coupled to the aircraft engine, and connected electrically to an indicator mounted on the instrument panel. These two units are connected by a current-carrying cable. The generator transmits three-phase power to the synchronous motor in the indicator. The frequency of the transmitted power is proportional to the engine speed. Through use of the magnetic drag principle, the indicator furnishes an accurate indication of engine speed.

Tachometer generators are small compact units, generally availble in three types: (1) The pad, (2) the swivel-nut, and (3) the screw type. These names are derived from the kind of mounting used in attaching the generator to the engine. The pad-type tachometer generator (figure 12–36A) is constructed with an end shield designed to permit attachment of the generator to a flat plate on the engine frame, or accessory reduction gearbox, with four bolts. The swivel-nut tachometer generator is constructed with a mounting nut which is free to turn in respect to the rest of the instrument. This type of generator can be held stationary while the mounting nut is screwed into place. The screw-type tachometer generator (figure 12–36B) is constructed with a mounting nut inserted in one of the generator end shields. The mounting nut is a rigid part of the instrument, and the whole generator must be turned to screw the nut onto its mating threads.

The dual tachometer consists of two tachometer indicator units housed in a single case. The indicator pointers show simultaneously on a single dial the r.p.m. of two engines. Some tachometer indicators are equipped with a flight-hour meter dial, usually located in the lower center area of the dial face, just below the pointer pivot.

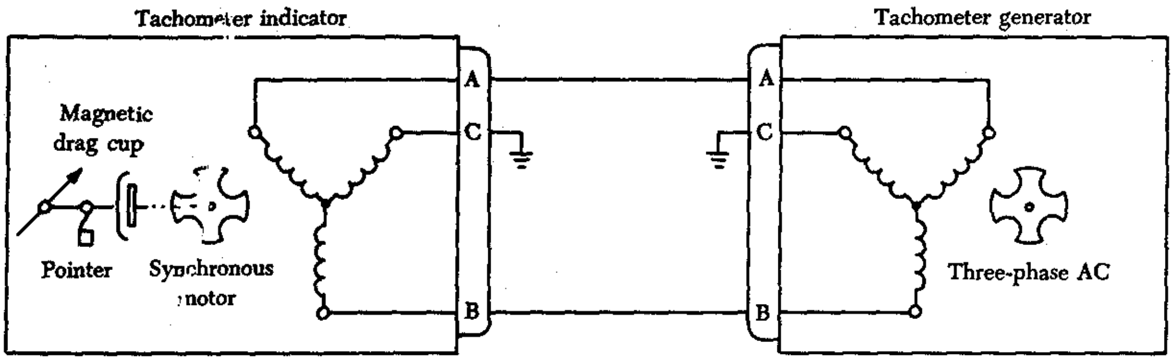
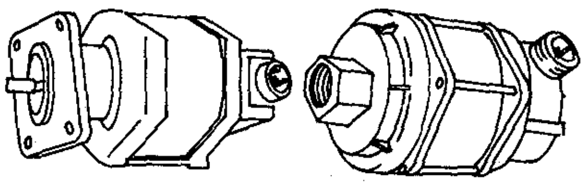


FIGURE 12–35. Schematic of a tachometer system.



A. Pad type B. Screw type

FIGURE 12–36. Tachometer generators.

Dual tachometers are also placed in the same case with a synchroscope for various purposes. One of these is the helicopter tachometer with synchroscope. This instrument shows simultaneously the speed of rotation of the engine crankshaft, the speed of rotation of the rotor shaft, and the slippage of the rotor due to malfunctioning of the clutch or excessive speed of the rotor when the clutch is disengaged in flight. The speed of both the rotor shaft and the engine shaft is indicated by a regular dual tachometer, and the slippage is indicated on a synchroscope (figure 12–37).

Tachometer Maintenance

Tachometer indicators should be checked for loose glass, chipped scale markings, or loose pointers. The difference in indications between readings taken before and after lightly tapping the instrument should not exceed approximately ±15 r.p.m. This value may vary, depending on the tolerance established by the indicator manufacturer. Both tachometer generator and indicator should be inspected for tightness of mechanical and electrical connections, security of mounting, and general condition. For detailed maintenance procedures, the manufacturer's instructions should always be consulted.

When an engine equipped with an electric tachometer is running at idle r.p.m. the tachometer indicator pointers may fluctuate and read low. This is an indication that the synchronous motor is not synchronized with the generator output. As the engine speed is increased the motor should synchronize and register the r.p.m. correctly. The r.p.m. at which synchronization occurs will vary with the design of the tachometer system.

If the instrument pointers oscillate at speeds above the synchronizing value, determine that the total oscillation does not exceed the allowable tolerance. If the oscillation exceeds the tolerance, determine if it is the instrument or one of the other components that is at fault.

Pointer oscillation can occur with a mechanical indicating system if the flexible drive is permitted to whip. The drive shaft should be secured at frequent intervals to prevent it from whipping.

When installing mechanical type indicators, be sure that the flexible drive has adequate clearance

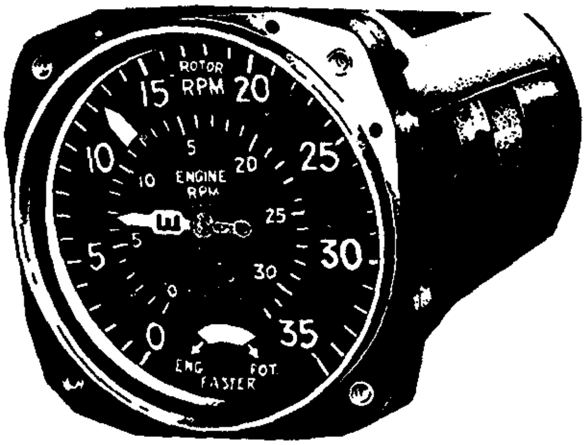


FIGURE 12–37. Helicopter tachometer with synchroscope.

behind the panel. Any bends necessary to route the drive should not cause strain on the instrument when it is secured to the panel. Avoid sharp bends in the drive; an improperly installed drive can cause the indicator to fail to read, or to read incorrectly.

SYNCHROSCOPE

The synchroscope is an instrument that indicates whether two (or more) engines are synchronized; that is, whether they are operating at the same r.p.m. The instrument consists of a small electric motor which receives electrical current from the tachometer generators of both engines. The synchroscope is designed so that current from the faster-running engine controls the direction in which the synchroscope motor rotates.

If both engines are operating at exactly the same speed, the synchroscope motor does not operate. If, however, one engine is operating faster than the other, its generator signal will cause the synchroscope motor to turn in a given direction. If the speed of the other engine then becomes greater than that of the first engine, the signal from its generator will then cause the synchroscope motor to reverse itself and turn in the opposite direction.

The motor of the synchroscope is connected by means of a shaft to a double-ended pointer on the dial of the instrument (figure 12–38). It is necessary to designate one of the two engines a master engine if the synchroscope indicators are to be useful. The dial readings, with leftward rotation of the pointer indicating "slow" and rightward motion indicating "fast," then refer to the operation of the second engine in relation to the speed of the master engine.

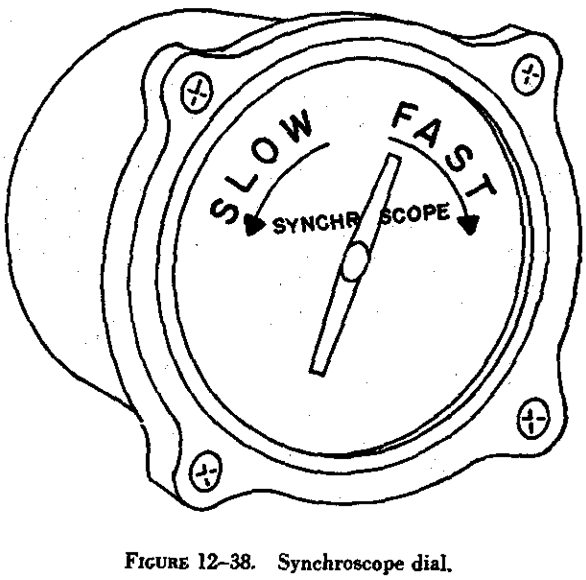


FIGURE 12–38. Synchroscope dial.

For aircraft with more than two engines, additional synchroscopes are used. One engine is designated the master engine, and synchroscopes are connected between its tachometer and those of each of the other individual engines. On a complete installation of this kind, there will, of course, be one less instrument than there are engines, since the master engine is common to all the pairs.

One type of four-engine synchroscope is a special instrument that is actually three individual synchroscopes in one case (figure 12–39).

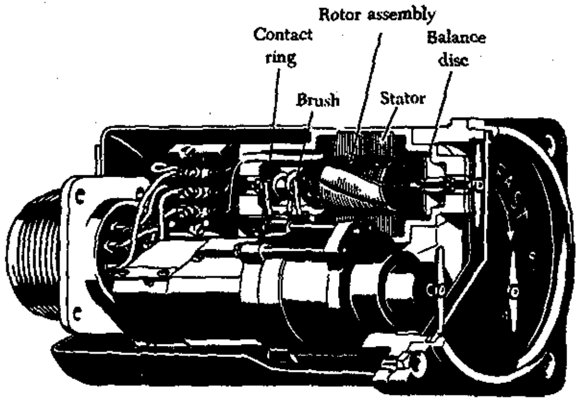


FIGURE 12–39. Four-engine synchroscope.

The rotor of each is electrically connected to the tachometer generator of the engine designated as the master, while each stator is connected to one of the other engine tachometers. There are three hands, each indicating the relative speed of the number two, three, or four engine, as shown in figure 12–40.

The separate hands revolve clockwise when their respective engine is running faster than the master and counterclockwise when it is running slower. Rotation of the hand begins as the speed difference reaches about 350 r.p.m., and as the engines approach synchronization the hand revolves at a ratio proportional to the speed difference.

TEMPERATURE INDICATORS

Various temperature indications must be known in order for an aircraft to be operated properly. It is important that the temperature of the engine oil, carburetor mixture, inlet air, free air, engine cylinders, heater ducts, and exhaust gas temperature of

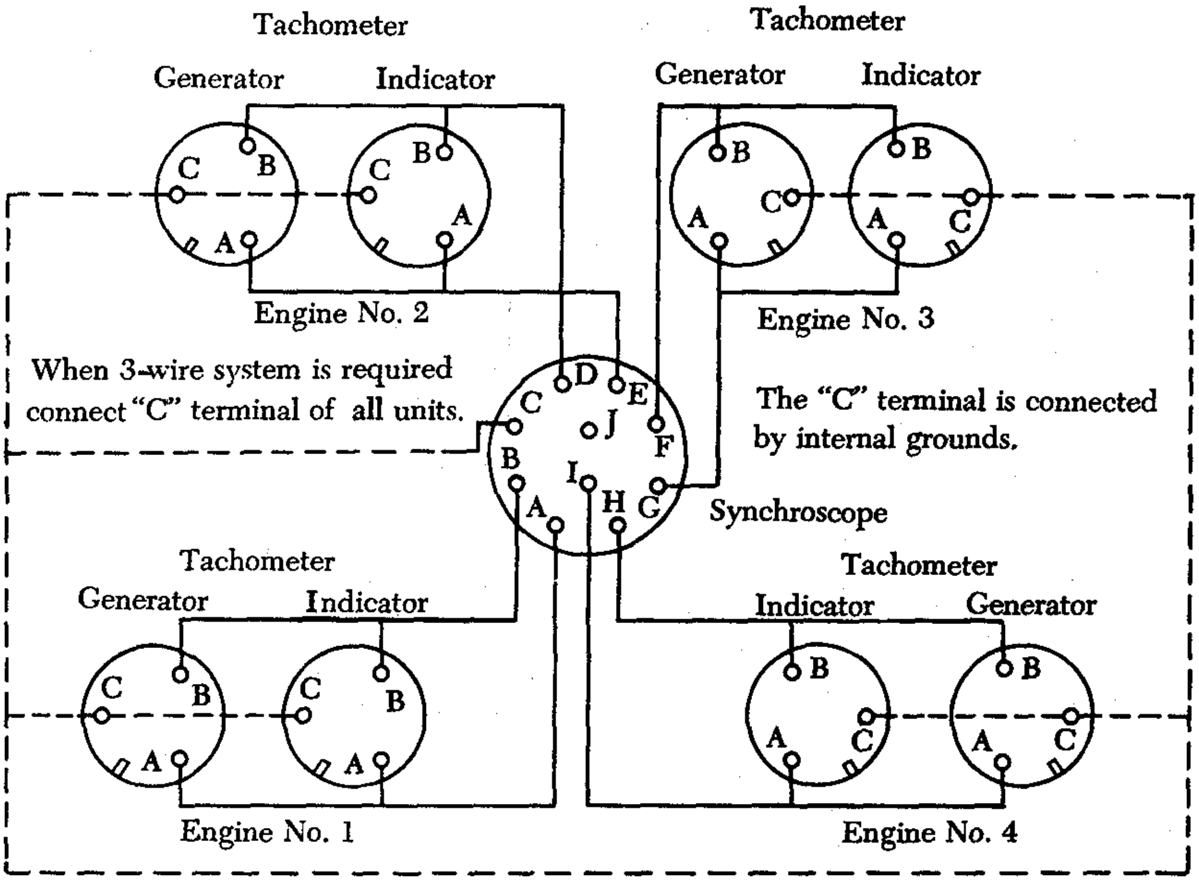


FIGURE 12–40. Four-engine synchroscope schematic.

turbine engines be known. Many other temperatures must also be known, but these are some of the more important. Different types of thermometers are used to collect and present this information.

Electrical Resistance Thermometer

Electrical resistance thermometers are used widely in many types of aircraft to measure carburetor air, oil, and free air temperatures.

The principal parts of the electrical resistance thermometer are the indicating instrument, the temperature-sensitive element (or bulb), and the connecting wires and plug connectors.

Oil temperature thermometers of the electrical resistance type have typical ranges of from —10° to +120° C., or from —70° to +150° C. Carburetor air and mixture thermometers may have a range of from —50° to +50° C., as do many free air thermometers.

A typical electrical resistance thermometer is shown in figure 12–41. Indicators are also available

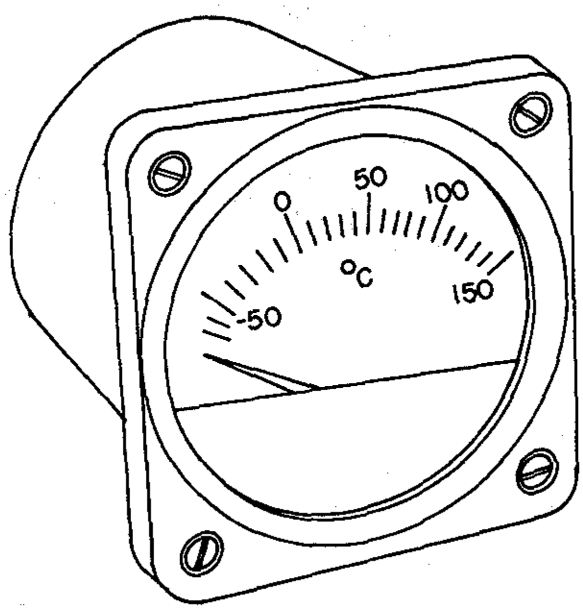


FIGURE 12–41. Typical electrical resistance temperature indicator.

in dual form for use in multi-engine aircraft. Most indicators are self-compensated for changes in cockpit temperature.

The electrical resistance thermometer operates on the principle of the change in the electrical resistance of most metals with changes in temperature. In most cases, the electrical resistance of a metal increases as the temperature rises. The resistance of some metals increases more than the resistance of others with a given rise in temperature. If a metallic resistor with a high temperature-resistant coefficient (a high rate of resistance rise for a given incresae in temperature) is subjected to a temperature to be measured, and a resistance indicator is connected to it, all the requirements for an electrical thermometer system are present.

The heat-sensitive resistor is the main element in the bulb. It is manufactured so that it has a definite resistance for each temperature value within its working range. The temperature-sensitive resistor element is a winding made of various alloys, such as nickel/manganese wire, in suitable insulating material. The resistor is protected by a closed-end metal tube attached to a threaded plug with a hexagon head (figure 12–42). The two ends of the winding are brazed or welded to an electrical receptacle designed to receive the prongs of the connector plug.

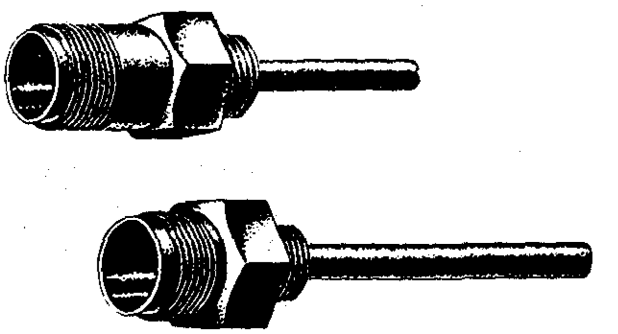


FIGURE 12–42. Two types of resistance thermometer bulb assemblies.

The electrical resistance indicator is a resistance-measuring instrument. Its dial is calibrated in degrees of temperature instead of ohms and measures temperature by using a modified form of the Wheatstone-bridge circuit.

The Wheatstone-bridge meter operates on the principle of balancing one unknown resistor against other known resistances. A simplified form of a Wheatstone-bridge circuit is shown in figure 12–43.

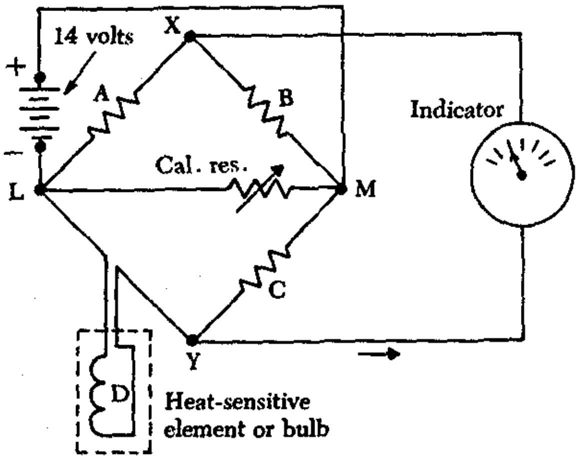


FIGURE 12–43. Wheatstone-bridge meter circuit.

Three equal values of resistances (A, B, and C, figure 12–43) are connected to a diamond-shaped bridge circuit with a resistance of unknown value (D).

The unknown resistance represents the resistance of the temperature bulb of the electrical resistance thermometer system. A galvanometer calibrated to read in degrees is attached across the circuit at point X and Y.

When the temperature causes the resistance of the bulb to equal that of the other resistances, no potential difference exists between points X and Y in the circuit, and no current flows in the galvanometer leg of the circuit. If the temperature of the bulb changes, its resistance will also change, and the bridge becomes unbalanced, causing current to flow through the galvanometer in one direction or the other.

The dial of the galvanometer is calibrated in degrees of temperature, converting it to a temperature-measuring instrument. Most indicators are provided with a zero adjustment screw on the face of the instrument to set the pointer at a balance point (the position of the pointer when the bridge is balanced and no current flows through the meter).

Thermocouple Thermometer Indicators

The cylinder temperature of most air-cooled reciprocating aircraft engines is measured by a thermometer which has its heat-sensitive element attached to some point on one of the cylinders (normally the hottest cylinder). In the case of turbojet engines, the exhaust temperature is measured by attaching thermocouples to the tailcone.

A thermocouple is a circuit or connection of two unlike metals; such a circuit has two junctions. If one of the junctions is heated to a higher temperature than the other, an electromotive force is produced in the circuit. By including a galvanometer in the circuit, this force can be measured. The hotter the high-temperature junction (hot junction) becomes, the greater the electromotive force produced. By calibrating the galvanometer dial in degrees it becomes a thermometer.

A typical thermocouple thermometer system (figure 12–44) used to indicate engine temperature consists of a galvanometer indicator calibrated in degrees of centigrade, a thermocouple, and thermocouple leads.

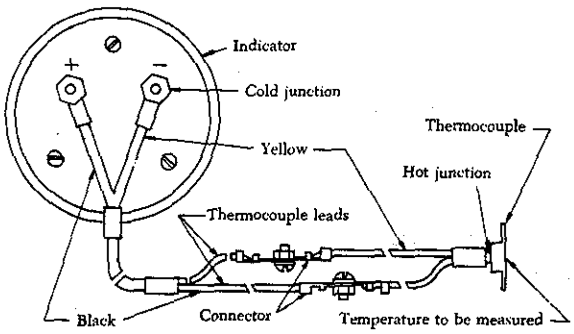


FIGURE 12–44. Reciprocating engine cylinder head temperature thermocouple system.

Thermocouple leads are commonly made from iron and constantan, but copper and constantan or chromel and alumel are other combinations of dissimilar metals in use. Iron/constantan is used mostly in radial engines, and chromel/alumel is used in jet engines.

Thermocouple leads are designed to provide a definite amount of resistance in the thermocouple circuit. Thus, their length or cross-sectional size cannot be altered unless some compensation is made for the change in total resistance.

The hot junction of the thermocouple varies in shape depending on its application. Two common types are shown in figure 12–45; they are the gasket type and the bayonet type. In the gasket type, two rings of dissimilar metals are pressed together to form a spark plug gasket. Each lead that makes a connection back to the galvanometer must be made of the same metal as the part of the thermocouple to which it is connected. For example, a copper wire is connected to the copper ring and a constantan wire is connected to the constantan ring. The bayonet type thermocouple fits into a hole or well in the cylinder head. Here again, the same metal is used in the lead as in the part of the thermocouple to which it is connected.

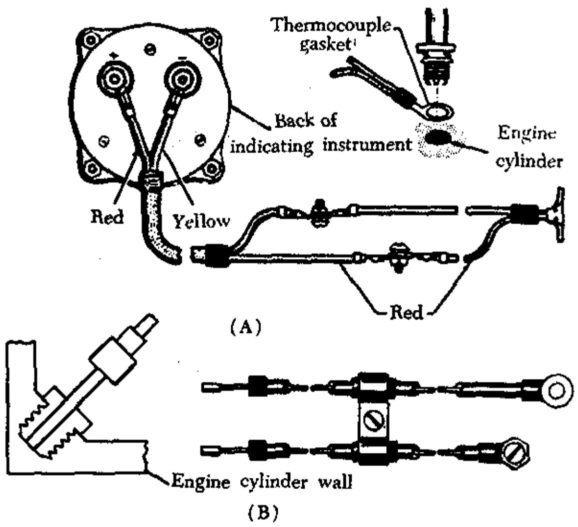


FIGURE 12–45. Thermocouples: (A) Gasket type, (B) Bayonet type.

The cylinder chosen for installing the thermocouple is the one which runs the hottest under most operating conditions. The location of this cylinder varies with different engines.

The cold junction of the thermocouple circuit is inside the instrument case.

Since the electromotive force set up in the circuit varies with the difference in temperature between the "hot" and "cold" junctions, it is necessary to compensate the indicator mechanism for changes in cockpit temperature which affect the "cold" junction. This is accomplished by using a bimetallic spring connected to the indicator mechanism.

When the leads are disconnected from the indicator, the temperature of the cockpit area around the instrument panel can be read on the indicator dial. This is because the bimetallic compensator spring continues to function as a thermometer.

Figure 12–46 shows the dials of two thermocouple temperature indicators.

Gas Temperature Indicating Systems

EGT (exhaust gas temperature) is a critical variable of turbine engine operation. The EGT indicating system provides a visual temperature indication in the cockpit of the turbine exhaust gases as they leave the turbine unit. In certain turbine engines the temperature of the exhaust gases is measured at

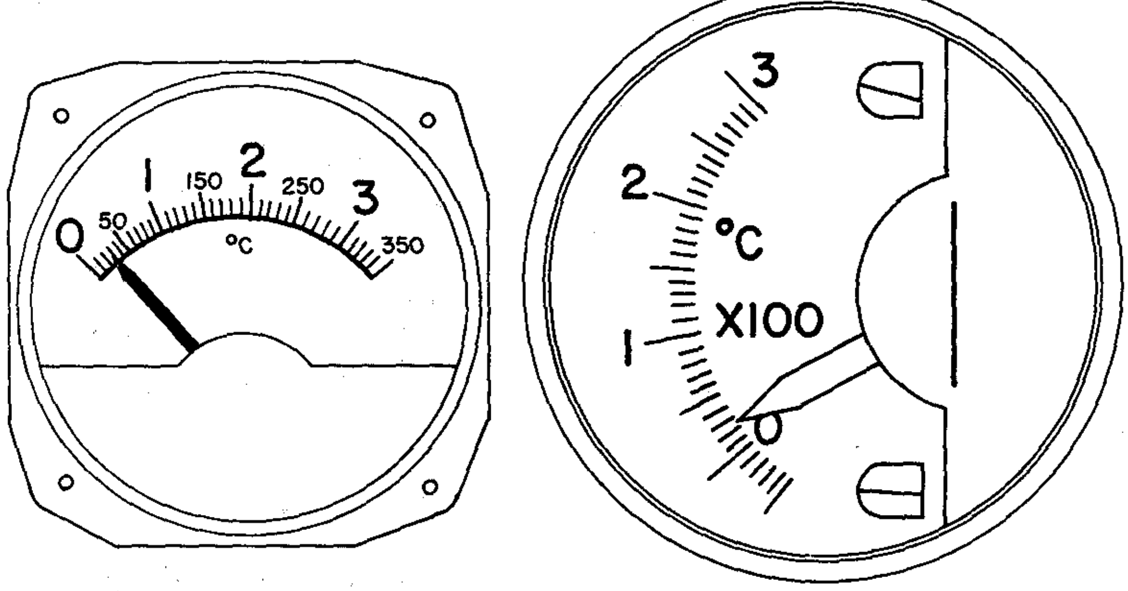


FIGURE 12–46. Two types of thermocouple temperature indicators.

the entrance to the turbine unit. This is usually referred to as a TIT (turbine inlet temperature) indicating system. The principal disadvantages of this method are that the number of thermocouples required become greater and the environmental temperatures in which they must operate are increased.

A gas temperature thermocouple is mounted in a ceramic insulator and encased in a metal sheath; the assembly forms a probe which projects into the exhaust gas stream. The thermocouple is made from chromel (a nickel/chromium alloy) and alumel (a nickel/aluminum alloy). The hot junction protrudes into a space inside the sheath. The sheath has transfer holes in the end of it which allow the exhaust gases to flow across the hot junction.

Several thermocouples are used and are spaced at intervals around the perimeter of the engine turbine casing or exhaust duct. The thermocouples measure engine EGT in millivolts, and this voltage is applied to the amplifier in the cockpit indicator, where it is amplified and used to energize the servomotor which drives the indicator pointer.

A typical EGT thermocouple system is shown in figure 12–47.

The EGT indicator shown is a hermetically sealed unit and has provisions for a mating electrical connector plug. The instrument's scale ranges from 0° C. to 1,200° C., with a vernier dial in the upper right-hand corner. A power "off" warning flag is located in the lower portion of the dial.

The TIT indicating system provides a visual indication at the instrument panel of the temperature of gases entering the turbine. On one type of turbine aircraft the temperature of each engine turbine inlet is measured by 18 dual-unit thermocouples installed in the turbine inlet casing. One set of these thermocouples is paralleled to transmit signals to the cockpit indicator. The other set of paralleled thermocouples provides temperature signals to the temperature datum control. Each circuit is electrically independent, providing dual system reliability.

The thermocouple assemblies are installed on pads around the turbine inlet case. Each thermocouple incorporates two electrically independent junctions within a sampling probe. The average voltage of the thermocouples at the thermocouple terminal blocks represents the TIT.

A schematic for the turbine inlet temperature system for one engine of a four-engine turbine aircraft is shown in figure 12–48. Circuits for the other three engines are identical to this system. The indicator contains a bridge circuit, a chopper circuit, a two-phase motor to drive the pointer, and a feedback potentiometer. Also included are a voltage reference circuit, an amplifier, a power "off" flag, a power supply, and an overtemperature warning light. Output of the amplifier energizes the variable field of the two-phase motor which positions the indicator main pointer and a digital indicator. The motor also drives the feedback potentiometer to pro-

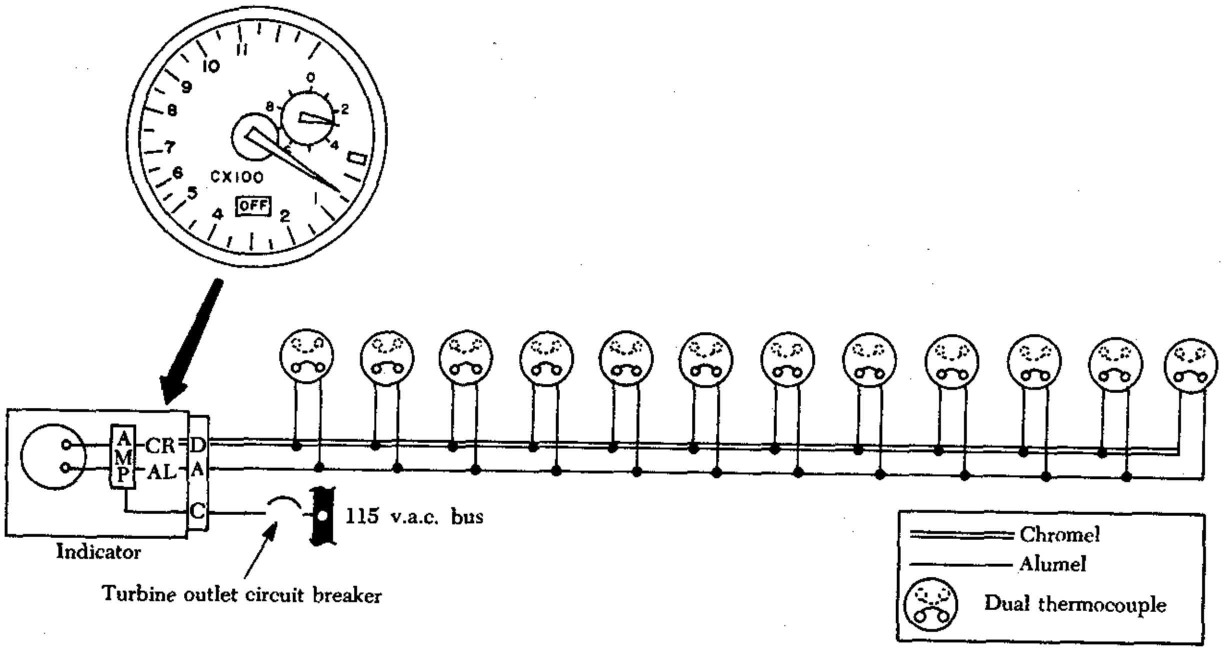


FIGURE 12–47. Typical exhaust gas temperature thermocouple system.

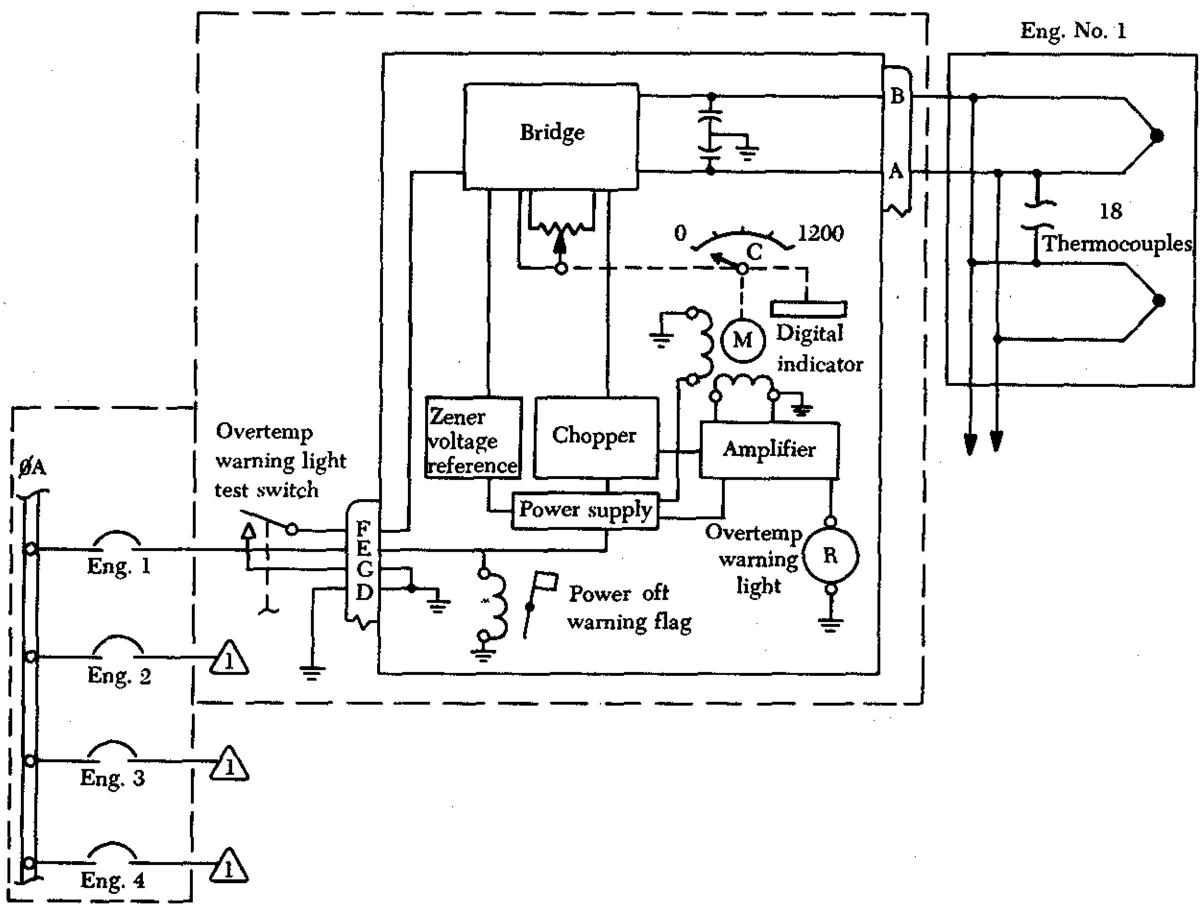


FIGURE 12–48. Turbine inlet temperature indicating system.

vide a huming signal to stop the drive motor when the correct pointer position, relative to the temperature signal, has been reached. The voltage reference circuit provides a closely regulated reference voltage in the bridge circuit to preclude error from input voltage variation to the indicator power supply.

The overtemperature warning light in the indicator illuminates when the TIT reaches a predetermined limit. An external test switch is usually installed so that overtemperature warning lights for all the engines can be tested at the same time. When the test switch is operated, an overtemperature signal is simulated in each indicator temperature control bridge circuit.

RATIOMETER ELECTRICAL RESISTANCE THERMOMETER

The basic Wheatstone-bridge, temperature-indicating system provides accurate indications when the pointer is at the balance point on the indicator dial. As the pointer moves away from the balance point, the Wheatstone-bridge indicator is increasingly affected by supply voltage variations. Greater accuracy can be obtained by inserting one of several types of automatic line voltage compensating circuits into the circuit. Some of these voltage regulators employ the filament resistance of lamps to achieve a more uniform supply voltage. The resistance of the lamp filaments helps regulate the voltage applied to the Wheatstone-bridge circuit since the filament resistance changes in step with supply voltage variation.

The ratiometer is a more sophisticated arrangement for obtaining greater accuracy in resistance-bulb indicators. The ratiometer measures the ratio of currents, using an adaptation of the basic Wheatstone-bridge with ratio circuitry for increased sensitivity.

A schematic of a ratiometer temperature circuit is shown in figure 12–49. The circuit contains two parallel branches, one with a fixed resistance in series with coil A, and the other a built-in resistance in series with coil B. The two coils are wound on a rotor pivoted in the center of the magnet air gap. The permanent magnet is arranged to provide a larger air gap between the magnet and the coils at the bottom than at the top. This produces a flux density that is progressively stronger from the bottom of the air gap to the top.

The direction of the current through each coil in respect to the polarity of the permanent magnet

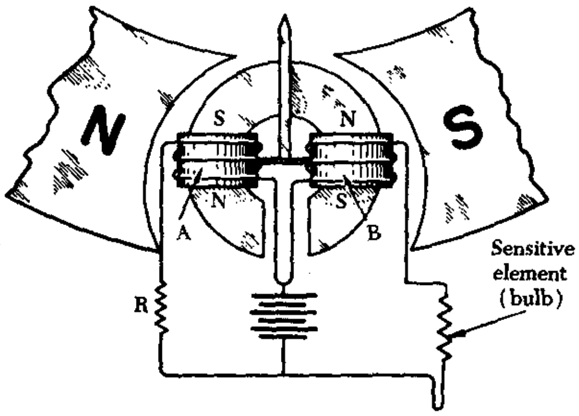


FIGURE 12–49. Ratiometer temperature-measuring system schematic.

causes the coil with the greater current flow to react in the weaker magnetic field. If the resistance of the temperature bulb is equal to the value of the fixed resistance, and equal values of current are flowing through the coils, the torque on the coils will be the same and the indicator points will be in the vertical (zero) position.

If the bulb temperature increases, its resistance will also increase, causing the current through the coil B circuit branch to decrease. Consequently, the torque on coil B decreases and coil A pushes downward into a weaker magnetic field; coil A, with its weaker current flow, moves into a stronger magnetic field. The torques on the coils still balance since the product of current times flux remains the same for both coils, but the pointer has moved to a new position on the calibrated scale. Just the opposite of this action would take place if the temperature of the heat-sensitive bulb should decrease.

Ratiometer temperature-measuring systems are used to measure engine oil, outside air, and carburetor air temperatures in many types of aircraft. They are especially in demand to measure temperature conditions where accuracy is important or large variations of supply voltages are encountered.

FUEL FLOWMETER SYSTEMS

Fuel flowmeter systems are used to indicate fuel usage. They are most commonly installed on large multi-engine aircraft, but they may be found on any type of aircraft if fuel economy is an important factor.

A typical flowmeter system for a reciprocating engine consists of a flowmeter transmitter and an indicator. The transmitter is usually connected into the fuel line leading from the carburetor outlet to the fuel feed valve or discharge nozzle. The indicator is normally mounted in the instrument panel.

A cross sectional view of a typical transmitter fuel chamber is shown in figure 12–50. Fuel entering the inlet side of the fuel chamber is directed against the metering vane, causing the vane to swing on its shaft within the chamber. As the vane is moved from a closed position by the pressure of the fuel flow, the clearance between the vane and the fuel chamber wall becomes increasingly larger.

Figure 12–51 shows an exploded view of a fuel flowmeter system. Note that the metering vane moves against the opposing force of a hairspring. When the force created by a given fuel flow is balanced by spring tension, the vane becomes stationary. The vane is connected magnetically to the rotor of a transmitter, which generates electrical signals to position the cockpit indicator. The distance the metering vane moves is proportional to, and a measure of, the rate of fuel flow.

The damper vane of the transmitter cushions fluctuations caused by air bubbles. The relief valve bypasses fuel to the chamber outlet when the flow of fuel is greater than chamber capacity.

A simplified schematic of a vane-type flowmeter system (figure 12–52) shows the metering vane connected to the flowmeter transmitter and the rotor and stator of the indicator connected to a common power source with the transmitter.

The dial of a fuel-flow indicator is shown in figure 12–53. Some fuel-flow indicators are calibrated in gallons per hour, but most of them indicate the measurement of fuel flow in pounds.

The fuel flowmeter system used with turbine engine aircraft is usually a more complex system than that used in reciprocating engine aircraft.

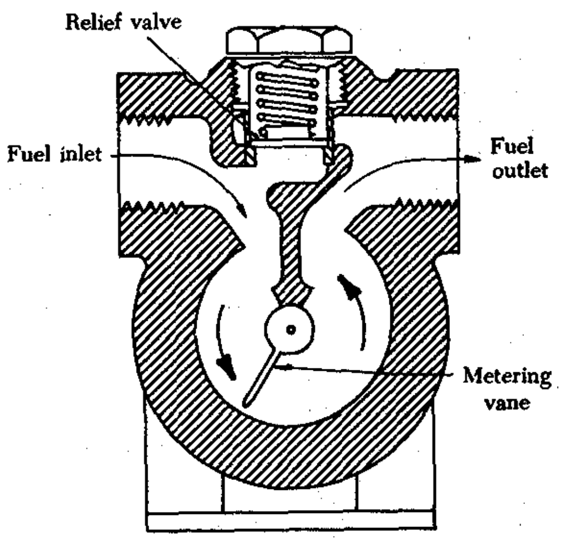


Figure 12–50. Flowmeter fuel chamber.

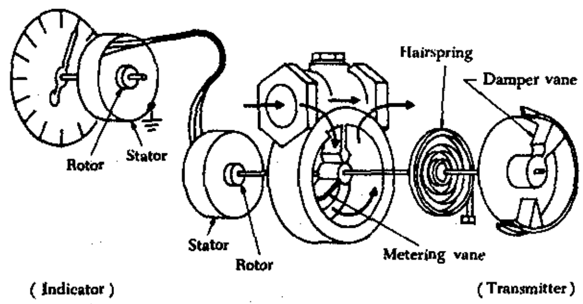


Figure 12–51. Fuel flowmeter system.

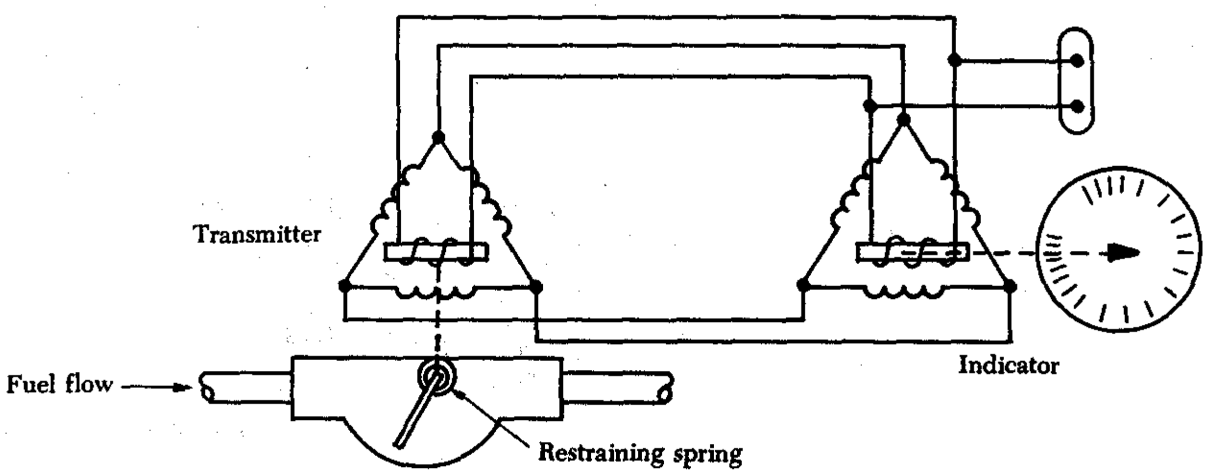


Figure 12–52. Schematic of vane-type flowmeter system.

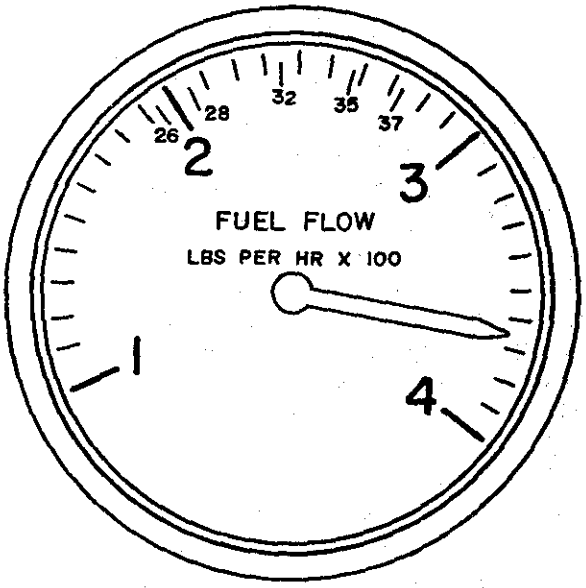


FIGURE 12-53. Typical fuel-flow indicator.

In the system shown schematically in figure 12-54, two cylinders, an impeller, and a turbine are mounted in the main fuel line leading to the engine. The impeller is driven at a constant speed by a special three-phase motor. The impeller imparts an angular momentum to the fuel, causing the turbine to rotate until the calibrated restraining spring force balances the force due to the angular momentum of the fuel. The deflection of the turbine positions the permanent magnet in the position transmitter to a position corresponding to the fuel flow in the line. This turbine position is transmitted electrically to the indicator in the cockpit.

GYROSCOPIC INSTRUMENTS

Three of the most common flight instruments, the attitude indicator, heading indicator, and the turn needle of the turn-and-bank indicator, are controlled by gyroscopes. To understand how these instruments operate requires a knowledge of gyroscopic principles, instrument power systems, and the operating principles of each instrument.

A gyroscope is a wheel or disk mounted to spin rapidly about an axis, and is also free to rotate about one or both of two axes perpendicular to

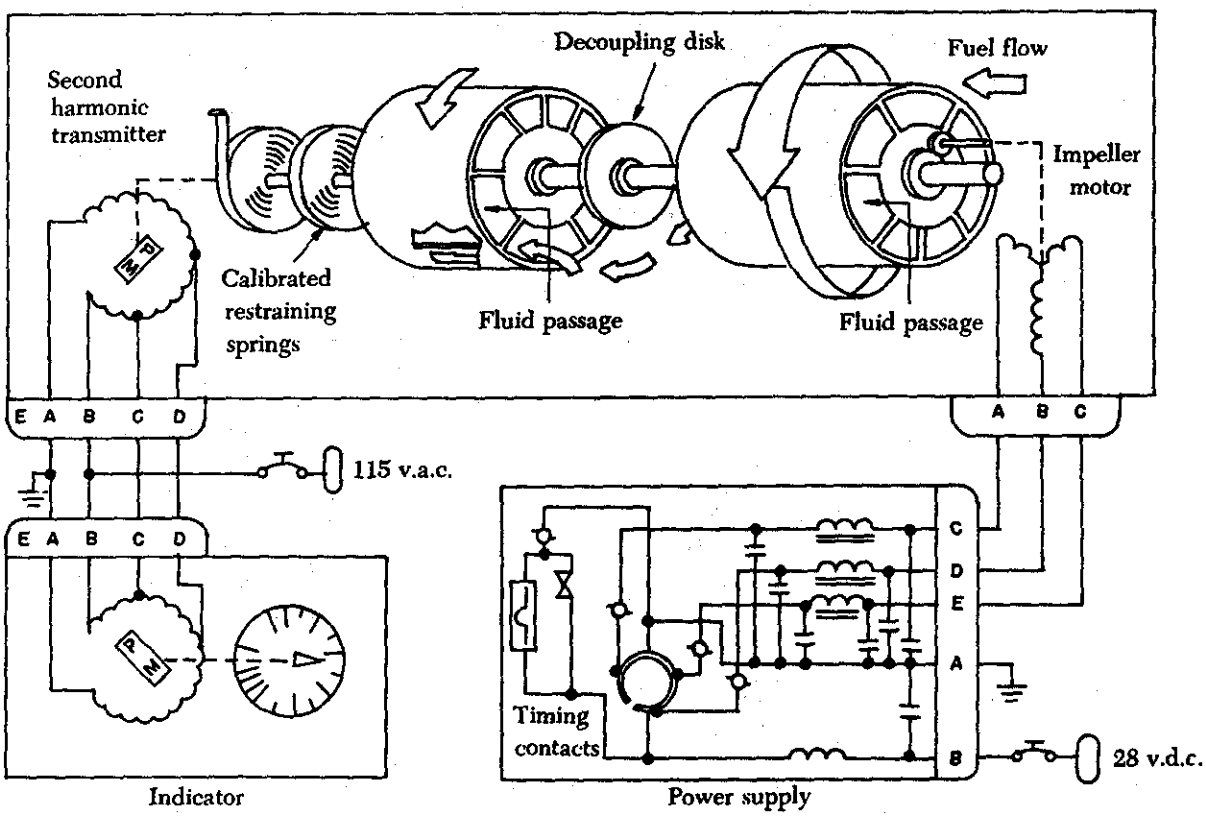


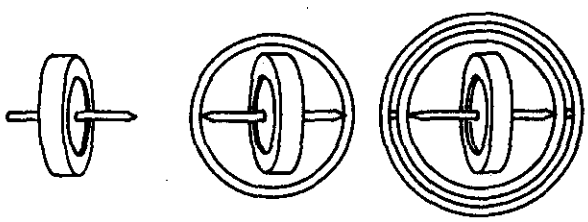
FIGURE 12-54. Schematic of a large turbine engine flowmeter system.

each other and to the axis of spin. A spinning gyroscope offers resistance to any force which tends to change the direction of the axis of spin.

A rotor and axle are the heart of a basic gyro (A of figure 12–55); a supporting ring with bearings on which the rotor and its axle can revolve are added to the basic unit (B of figure 12–55); and an outer ring with bearings at 90° to the rotor bearings has been added (C of figure 12–55). The inner ring with its rotor and axle can turn through 360° inside this outer ring.

A gyro at rest is shown in six different positions (figure 12–56) to demonstrate that unless the rotor is spinning a gyro has no unusual properties; it is simply a wheel universally mounted.

When the rotor is rotated at a high speed, the gyro exhibits one of its two gyroscopic characteristics. It acquires a high degree of rigidity, and its axle points in the same direction no matter how much its base is turned about (figure 12–57).



A B C

FIGURE 12–55. Basic gyroscope.

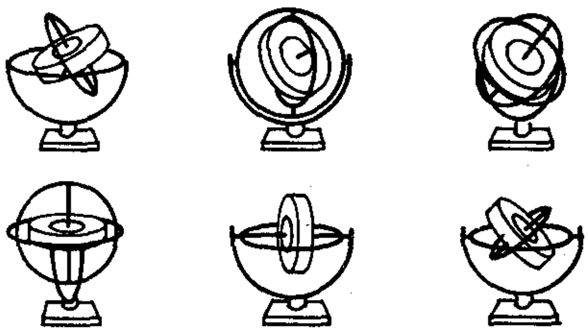


FIGURE 12–56. A gyro at rest.

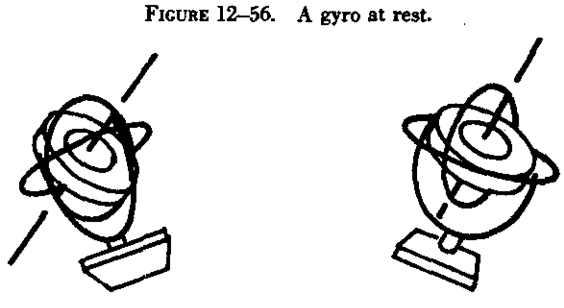


FIGURE 12–57. Gyroscope inertia.

Gyrocsopic rigidity depends upon several design factors:

(1) **Weight.** For a given size, a heavy mass is more resistant to disturbing forces than a light mass.
(2) **Angular velocity.** The higher the rotational speed, the greater the rigidity or resistance to deflection.
(3) **Radius at which the weight is concentrated.** Maximum effect is obtained from a mass when its principal weight is concentrated near the rim rotating at high speed.
(4) **Bearing friction.** Any friction applies a deflecting force to a gyro. Minimum bearing friction keeps deflecting forces at a minimum.

A second gyroscopic characteristic, precession, is illustrated in figure 12–58A by applying a force or pressure to the gyro about the horizontal axis. The applied force is resisted, and the gyro, instead of turning about its horizontal axis, turns or "precesses" about its vertical axis in the direction indicated by the letter P. In a similar manner, if pressure is applied to the vertical axis, the gyro will precess about its horizontal axis in the direction shown by the arrow P in figure 12–58B.

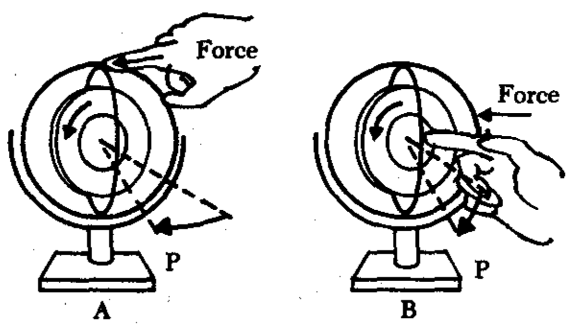


FIGURE 12–58. Gyrosocopic precession.

Two types of mountings are used, depending upon how the gyroscopic properties are to be used in the operation of an instrument. A freely or universally mounted gyro is set on three gimbals (rings), with the gyro free to rotate in any plane. Regardless of the position of the gyro base, the gyro tends to remain rigid in space. In the attitude indicator the horizon bar is gyro-controlled to remain parallel to the natural horizon, and changes in position of the aircraft are shown pictorially on the indicator.

The semirigid, or restricted, mounting employs two gimbals, limiting the rotor to two planes of rotation. In the turn-and-bank indicator, the semirigid mounting provides controlled precession of the rotor, and the precessing force exerted on the gyro by the turning aircraft causes the turn needle to indicate a turn.

SOURCES OF POWER FOR GYRO OPERATION

The gyroscopic instruments can be operated either by a vacuum system or an electrical power source. In some aircraft, all the gyros are either vacuum or electrically motivated; in others, vacuum (suction) systems provide the power for the attitude and heading indicators, while the electrical system drives the gyro for operation of the turn needle. Either alternating or direct current is used to power the gyroscopic instruments.

Vacuum System

The vacuum system spins the gyro by sucking a stream of air against the rotor vanes to turn the rotor at high speed, essentially as a water wheel or turbine operates. Air at atmospheric pressure drawn through a filter or filters drives the rotor vanes, and is sucked from the instrument case through a line to the vauuum source and vented to the atmosphere. Either a venturi or a vacuum pump can be used to provide the vacuum required to spin the rotors of the gyro instruments.

The vacuum value required for instrument operation is usually between 3½ in. to 4½ in. Hg and is usually adjusted by a vacuum relief valve located in the supply line. The turn-and-bank indicators used in some installations require a lower vacuum setting. This is obtained using an additional regulating valve in the individual instrument supply line.

Venturi-Tube Systems

The advantages of the venturi as a suction source are its relatively low cost and simplicity of installation and operation. A light, single-engine aircraft can be equipped with a 2-in. venturi (2 in. Hg vacuum capacity) to operate the turn needle. With an additional 8-in. venturi, power is available for the attitude and heading indicators. A venturi vacuum system is shown in figure 12–59.

The line from the gyro (figure 12–59) is connected to the throat of the venturi mounted on the exterior of the aircraft fuselage. Throughout the normal operating airspeed range the velocity of the

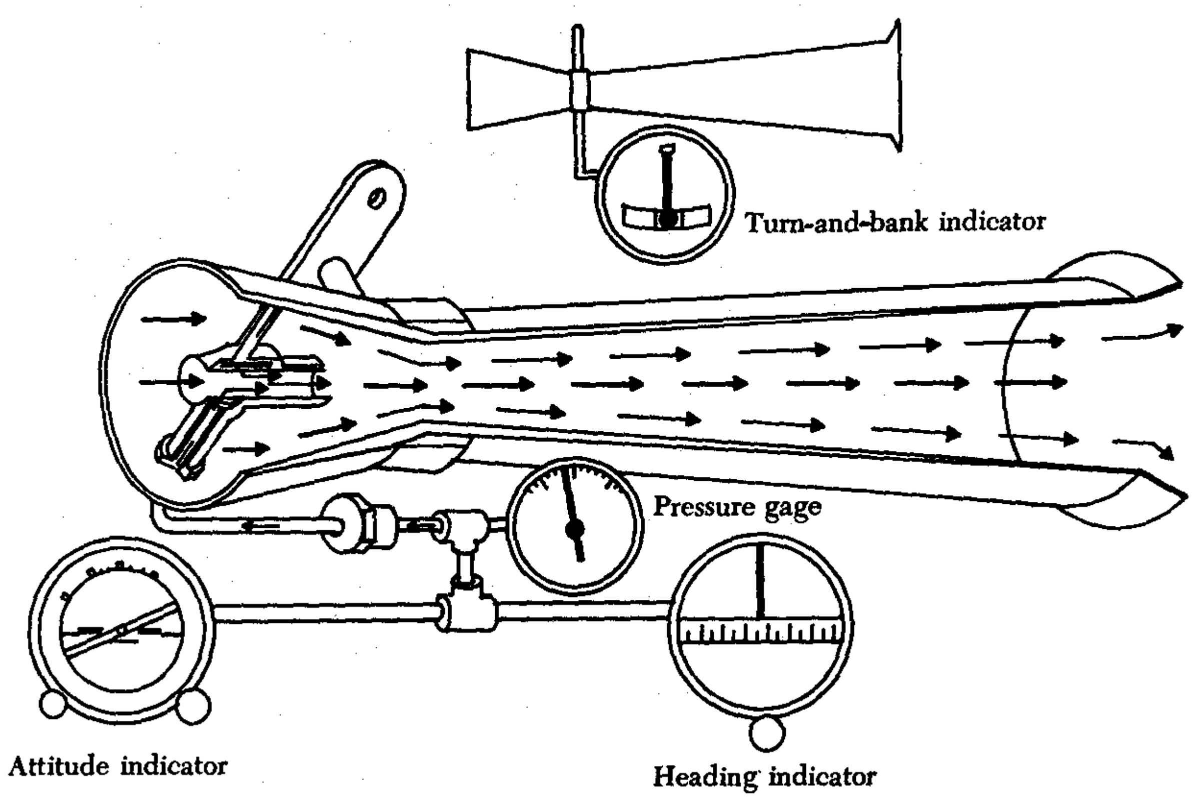


FIGURE 12–59. Venturi vacuum system.

air through the venturi creates sufficient suction to spin the gyro.

The limitations of the venturi system should be evident from the illustration in figure 12–59. The venturi is designed to produce the desired vacuum at approximately 100 m.p.h. under standard sea-level conditions. Wide variations in airspeed or air density, or restriction to airflow by ice accretion, will affect the pressure at the venturi throat and thus the vacuum driving the gyro rotor. And, since the rotor does not reach normal operating speed until after takeoff, preflight operational checks of venturi-powered gyro instruments cannot be made. For this reason the system is adequate only for light-aircraft instrument training and limited flying under instrument weather conditions. Aircraft flown throughout a wider range of speed, altitude, and weather conditions require a more effective source of power independent of airspeed and less susceptible to adverse atmospheric conditions.

Engine-Driven Vacuum Pump

The vane-type engine-driven pump is the most common source of vacuum for gyros installed in general aviation light aircraft. One type of engine-driven pump is mounted on the accessory drive shaft of the engine, and is connected to the engine lubrication system to seal, cool, and lubricate the pump.

Another commonly used source of vacuum is the dry vacuum pump, also engine-driven. The pump operates without lubrication, and the installation requires no lines to the engine oil supply, and no air-oil separator or gate check valve. In other respects, the dry pump system and oil lubricated system are the same.

The principal disadvantage of the pump-driven vacuum system relates to erratic operation in high-altitude flying. Apart from routine maintenance of the filters and plumbing, which are absent in the electric gyro, the engine-driven pump is as effective a source of power for light aircraft as the electrical system.

Typical Pump-Driven Vacuum System

Figure 12–61 shows the components of a vacuum system with a pump capacity of approximately 10" Hg at engine speeds above 1000 rpm. Pump capacity and pump size vary in different aircraft, depending on the number of gyros to be operated.

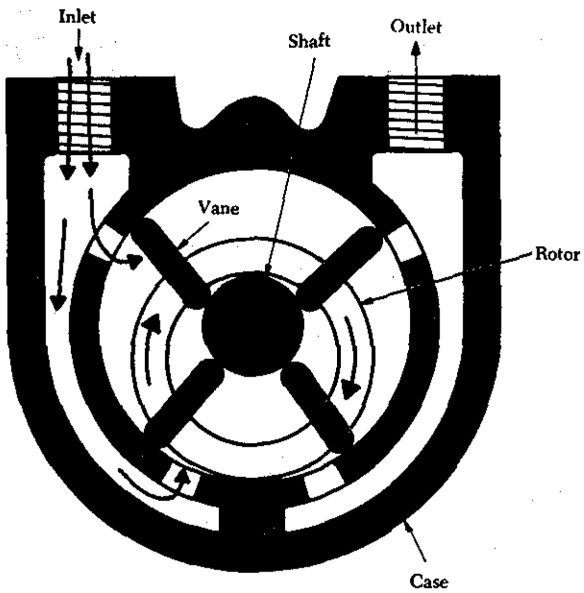


FIGURE 12–60. Cutaway view of a vane-type engine-driven vacuum pump.

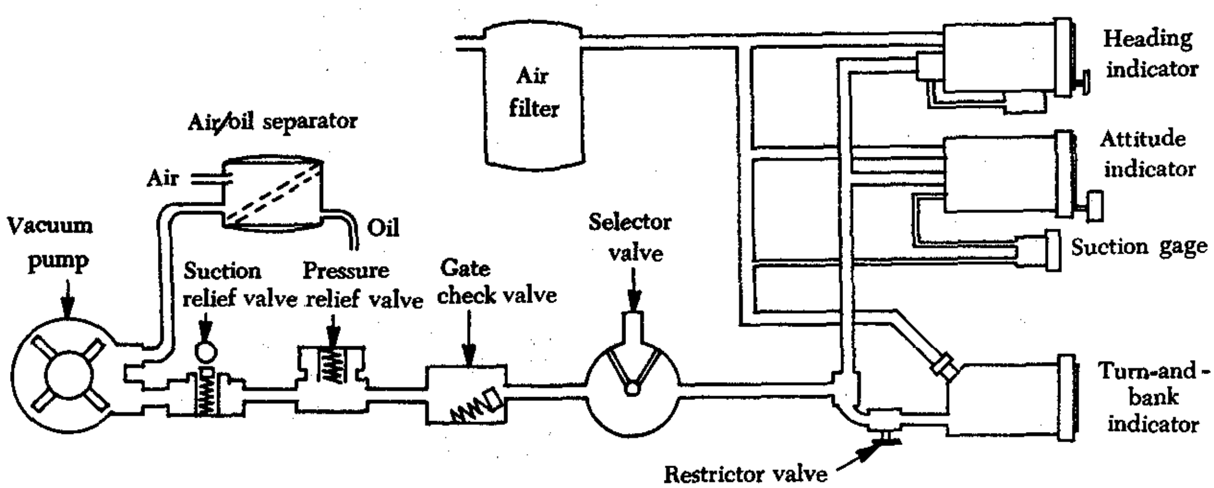


FIGURE 12–61. Typical pump-driven vacuum system.

Air-Oil Separator.—Oil and air in the vacuum pump are exhausted through the separator, which separates the oil from the air; the air is vented outboard, and the oil is returned to the engine sump.

Suction Relief Valve.—Since the system capacity is more than is needed for operation of the instruments, the adjustable suction relief valve is set for the vacuum desired for the instruments. Excess suction in the instrument lines is reduced when the spring-loaded valve opens to atmospheric pressure. (See fig. 12–62.)

Pressure Relief Valve.—Since a reverse flow of air from the pump would close both the gate check valve and the suction relief valve, the resulting pressure could rupture the lines. The pressure relief valve vents positive pressure into the atmosphere.

Gate Check Valve.—The gate check valve prevents possible damage to the instruments by engine back-fire, which would reverse the flow of air and oil from the pump. (See fig. 12–63.)

Selector Valve.—In twin-engine aircraft having vacuum pumps driven by both engines, the alternate pump can be selected to provide vacuum in the event of either engine or pump failure, with a check valve incorporated to seal off the failed pump.

Restrictor Valve.—Since the turn needle operates on less vacuum than that required for other gyro instruments, the vacuum in the main line must be reduced. This valve is either a needle valve adjusted to reduce the vacuum from the main line by approximately one-half, or a spring-loaded regulating valve that maintains a constant vacuum for the turn indicator, unless the main line vacuum falls below a minimum value.

Air Filter.—The master air filter screens foreign matter from the air flowing through all the gyro instruments, which are also provided with individual filters. Clogging of the master filter will reduce airflow and cause a lower reading on the suction gage. In aircraft having no master filter installed, each instrument has its own filter. With an individual filter system, clogging of a filter will not necessarily show on the suction gage.

Suction Gage.—The suction gage is a pressure gage, indicating the difference in inches of mercury, between the pressure inside the system and atmospheric or cockpit pressure. The desired vacuum, and the minimum and maximum limits, vary with gyro design. If the desired vacuum for the attitude and heading indicators is 5″ and the minimum in 4.6″, a reading below the latter value indicates that the airflow is not spinning the gyros fast enough for reliable operation. In many aircraft, the system provides a suction gage selector valve, permitting the pilot to check the vacuum at several points in the system.

Suction

Suction pressures discussed in conjunction with the operation of vacuum systems are actually minus or negative pressures (below sea level). For example, if sea level equals 17.5 p.s.i. then 1″ Hg (1 inch mercury) or 1 p.s.i. vacuum is equal to −1 p.s.i. negative pressure or 16.5 positive pressure. Likewise 3″ Hg = −3 p.s.i. negative pressure or +14.5 positive pressure.

Of course, for every action there is an equal and opposite reaction. Therefore when the vacuum pump develops a vacuum (negative pressure) it must also create pressure (positive). This pressure (compressed air) is sometimes utilized to operate pressure instruments, deicer boots and inflatable seals.

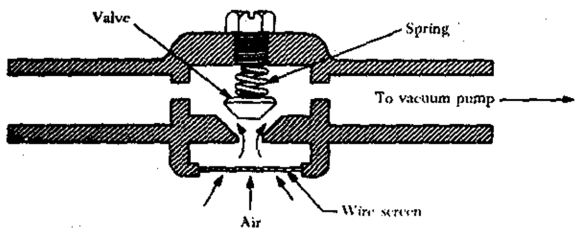


Figure 12–62. Vacuum regulator valve.

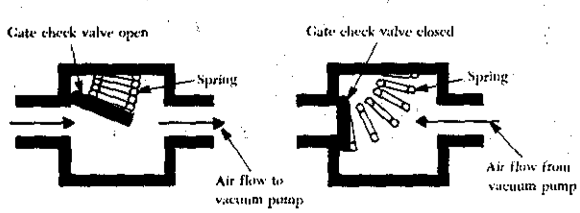


Figure 12–63. Gate check valve.

Typical System Operation

The schematic of a vacuum system for a twin-engine aircraft is shown in figure 12–64. This vacuum system consists of the following components; two engine-driven pumps, two vacuum relief valves, two flapper-type check valves, a vacuum manifold, a vacuum restrictor for each turn-and-bank indicator, an engine four-way selector valve, one vacuum gage, and a turn-and-bank selector valve.

The left and right engine-driven vacuum pumps and their associated lines and components are isolated from each other, and act as two independent vacuum systems. The vacuum lines are routed from each vacuum pump through a vacuum relief valve and through a check valve to the vacuum four-way selector valve.

From the engine four-way selector valve, which permits operation of the left or right engine vacuum system, the lines are routed to a vacuum manifold. From the manifold, flexible hose connects the vacuum-operated instruments into the system. From the instrument, lines routed to the vacuum gage pass through a turn-and-bank selector valve. This valve has three positions: main, left T&B, and right T&B. In the main position the vacuum gage indicates the vacuum in the lines of the artificial horizon and directional gyros. In the other positions, the lower value of vacuum for the turn-and-bank indicators can be read.

VACUUM-DRIVEN ATTITUDE GYROS

In a typical vacuum-driven attitude gyro system, air is sucked through the filter, then through pas-

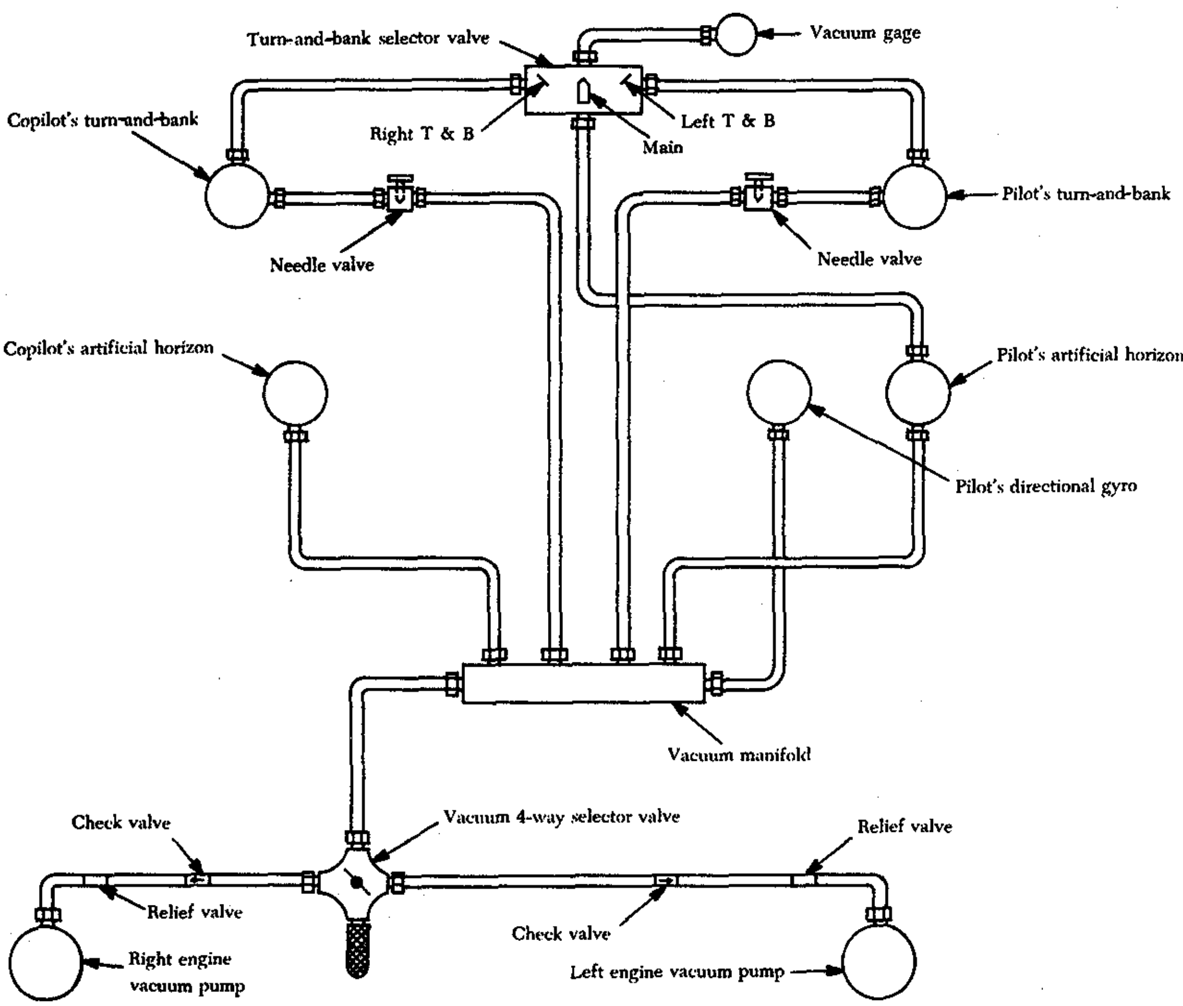


Figure 12–64. Vacuum system for a multi-engine aircraft.

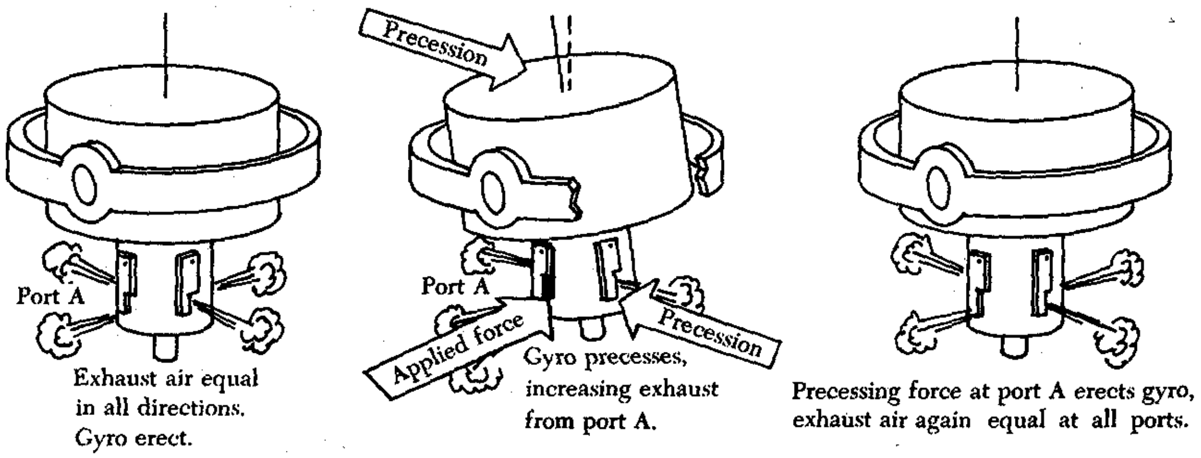


FIGURE 12–65. Erecting mechanism of a vacuum-driven attitude indicator.

sages in the rear pivot and inner gimbal ring, then into the housing where it is directed against the rotor vanes through two openings on opposite sides of the rotor. The air then passes through four equally spaced ports in the lower part of the rotor housing and is sucked out into the vacuum pump or venturi (figure 12–65).

The chamber containing the ports is the erecting device that returns the spin axis to its vertical alignment whenever a precessing force, such as bearing friction, displaces the rotor from its horizontal plane. Four exhaust ports are each half-covered by a pendulous vane, which allows discharge of equal volumes of air through each port when the rotor is properly erected. Any tilting of the rotor disturbs the total balance of the pendulous vanes, tending to close one vane of an opposite pair while the opposite vane opens a corresponding amount. The increase in air volume through the opening port exerts a precessing force on the rotor housing to erect the gyro, and the pendulous vanes return to a balanced condition (figure 12–66).

The limits of the attitude indicator specified in the manufacturer's instructions refer to the maximum rotation of the gimbals beyond which the gyro will tumble. The bank limits of a typical vacuum-driven attitude indicator are from approximately 100° to 110°, and the pitch limits vary from approximately 60° to 70°, depending on the design of a specific unit. If, for example, the pitch limits are 60° with the gyro normally erected, the rotor will tumble when the aircraft climb or dive angle exceeds 60°. As the rotor gimbal hits the stops, the

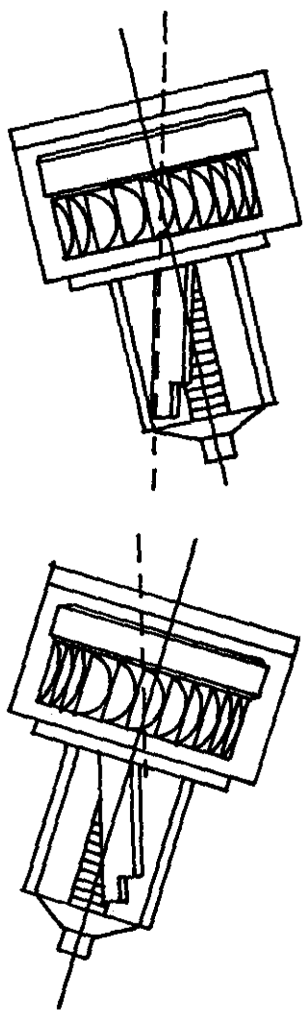


FIGURE 12–66. Action of pendulous vanes.

rotor precesses abruptly, causing excessive friction and wear on the gimbals. The rotor will normally precess back to the horizontal plane at a rate of approximately 8° per min.

Many gyros include a caging device, used to erect the rotor to its normal operating position prior to fight or after tumbling, and a flag to indicate that the gyro must be uncaged before use. Turning the caging knob prevents rotation of the gimbals and locks the rotor spin axis in its vertical position.

PRESSURE-OPERATED GYROS

The availability of pressure pumps on which no lubrication is necessary makes pressure-operated gyro systems feasible. In such installations, air is pushed under pressure through gyro instruments, rather than being sucked through the system. Positive-pressure pumps are more efficient than vacuum pumps, especially at higher altitudes.

VACUUM SYSTEM MAINTENANCE PRACTICES

Errors in the indications presented on the attitude indicator will result from any factor that prevents the vacuum system from operating within the design suction limits, or from any force that disturbs the free rotation of the gyro at design speed. These include poorly balanced components, clogged filters, improperly adjusted valves and pump malfunction. Such errors can be minimized by proper installation, inspection, and maintenance practices.

Other errors, inherent in the construction of the instrument, are caused by friction and worn parts. These errors, resulting in erratic precession and failure of the instrument to maintain accurate indications, increase with the service life of the instrument.

For the aviation mechanic the prevention or correction of vacuum system malfunctions usually consists of cleaning or replacing filters, checking

Possible Cause	*Isolation Procedure*	*Correction*
(1) *No Vacuum Pressure or Insufficient Pressure*		
Defective vacuum gage.	On multi-engine aircraft check opposite engine system on the gage.	Replace faulty vacuum gage.
Vacuum relief valve incorrectly adjusted.	Change valve adjustment.	Make final adjustment to proper setting of valve.
Vacuum relief valve installed backwards.	Visually inspect.	Install properly.
Broken line.	Visually inspect.	Replace line.
Lines crossed.	Visually inspect.	Install lines properly.
Obstruction in vacuum line.	Check for collapsed line.	Clean and test line. Replace defective part(s).
Vacuum pump failure.	Remove and inspect.	Replace faulty pump.
Vacuum regulator valve incorrectly adjusted.	Make valve adjustment and note pressure.	Adjust to proper pressure.
Vacuum relief valve dirty.	Clean and adjust relief valve.	Replace valve if adjustment fails.
(2) *Excessive Vacuum*		
Relief valve improperly adjusted.	----------------------------	Adjust relief valve to proper setting.
Inaccurate vacuum gage.	Check calibration of gage.	Replace faulty gage.
(3) *Gyro Horizon Bar Fails to Respond*		
Instrument caged.	Visually inspect.	Uncage instrument.
Instrument filter dirty.	Check filter.	Replace or clean as necessary.
Insufficient vacuum.	Check vacuum setting.	Adjust relief valve to proper setting.
Instrument assembly worn or dirty.	----------------------------	Replace instrument.
(4) *Turn-and-Bank Indicator Fails to Respond*		
No vacuum supplied to instrument.	Check lines and vacuum system.	Clean or replace lines and replace components of vacuum system as necessary.
Instrument filter clogged.	Visually inspect.	Replace filter.
Defective instrument.	Test with properly functioning instrument.	Replace faulty instrument.
(6) *Turn-and-Bank Pointer Vibrates*		
Defective instrument.	Test with properly functioning instrument	Replace defective instrument.

FIGURE 12-67. Vacuum system troubleshooting.

and correcting for insufficient vacuum, or removing and replacing the instruments. A list of the most common malfunctions, together with their correction, is included in figure 12–67.

ELECTRIC ATTITUDE INDICATOR

In the past, suction-driven gyros have been favored over the electric type for light aircraft because of their comparative simplicity and lower cost. However, the increasing importance of the attitude indicator has stimulated development of improved electric-driven gyros suited for light plane installation. Improvements relating to basic gyro design factors, easier readability, erection characteristics, reduction of induced errors, and instrument limitations are reflected in several available types. Depending upon the particular design improvements, the details for the instrument display and cockpit controls will vary among different instruments. All of them present, to a varying degree, the essential pitch-and-bank information for attitude reference.

The typical attitude indicator, or gyro horizon as it is sometimes referred to, has a vertical-seeking gyro, the axis of rotation tending to point toward the center of the earth. The gyro is linked with a horizon bar and stabilizes a kidney-shaped sphere having pitch attitude markings. The sphere, horizon bar, and bank index pointer move with changes of aircraft attitude. Combined readings of these presentations give a continuous pictorial presentation of the aircraft attitude in pitch and roll with respect to the earth's surface.

The gyroscope motor is driven by 115 v., 400 Hz alternating current. The gyro, turning at approximately 21,000 r.p.m., is supported by the yoke and pivot assembly (gimbals). Attached to the yoke and pivot assembly is the horizon bar, which moves up and down through an arc of approximately 27°. The kidney-shaped sphere provides a background for the horizon bar and has the words climb and dive and a bull's-eye painted on it. Climb and dive represent about 60° of pitch. Attached to the yoke and pivot assembly is the bank index pointer, which is free to rotate 360°. The dial face of the attitude indicator is marked with 0°, 10°, 20°, 30°, and 60° of bank, and is used with the bank index pointer to indicate the degree of bank left or right. The face of one type of gyro-horizon is shown in figure 12–68.

The function of the erection mechanism is to keep the gyro axis vertical to the surface of the earth. A magnet attached to the top of the gyro shaft spins at approximately 21,000 r.p.m. Around this magnet, but not attached, is a sleeve that is rotated by magnetic attraction at approximately 44 to 48 r.p.m. As illustrated in figure 12–69, the steel balls are free to move around the sleeve. If the pull of gravity is not aligned with the axis of the gyro, the balls will fall to the low side. The resulting precession re-aligns the axis of rotation vertically.

The gyro can be caged manually by a lever and cam mechanism to provide rapid erection. When the instrument is not getting sufficient power for normal operation, an "off" flag appears in the upper right-hand face of the instrument.

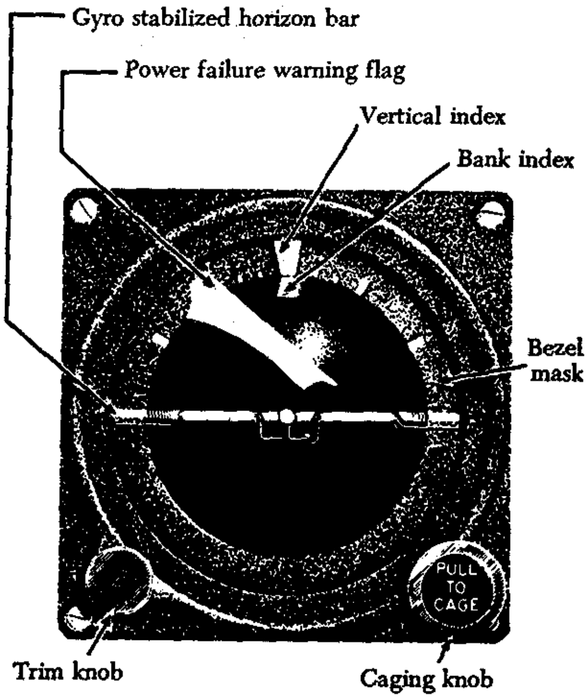


FIGURE 12–68. Gyro-horizon indicator.

Magnetic Compass

The magnetic compass is a simple, self-contained instrument which operates on the principle of magnetic attraction.

If a bar magnet is mounted on a pivot to be free to rotate in a horizontal plane, it will assume a position with one of its ends pointing toward the earth's north magnetic pole. This end of the magnet is called the north-seeking end of the magnet.

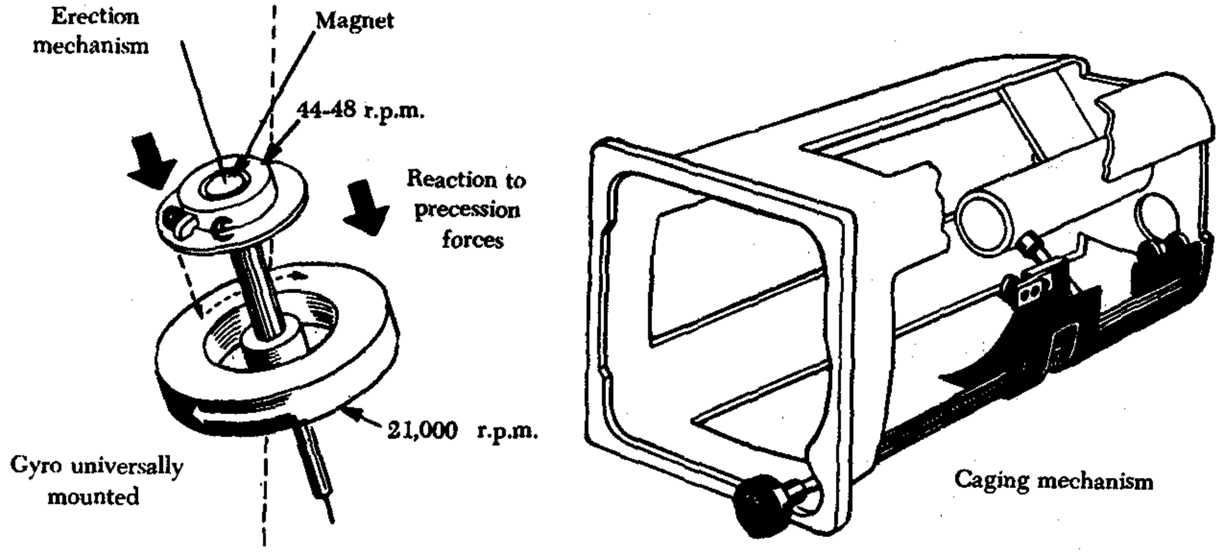


FIGURE 12-69. Erecting and caging mechanisms of an electric attitude indicator.

The magnetic compass consists of a liquid-filled bowl containing a pivoted float element to which one or more bar magnets, called needles, are fastened. The liquid in the bowl dampens the oscillations of the float and decreases the friction of the pivot. A diaphragm and vent provide for expansion and contraction of the liquid as altitude and/or temperatures change.

If more than one magnet is used in a compass, the magnets are mounted parallel to each other, with like poles pointing in the same direction. The element on which the magnets are mounted is so suspended that the magnets are free to align themselves with the earth's north and south magnetic poles.

A compass card, usually graduated in 5° increments, is attached to the float element of the compass. A fixed reference marker, called a lubber line, is attached to the compass bowl. The lubber line and the graduations on the card are visible through a glass window. The magnetic heading of the aircraft is read by noting the graduation on which the lubber line falls. The two views of a magnetic compass in figure 12-70 show the face and the internal components of a magnetic compass.

A compensating device containing small permanent magnets is incorporated in the compass to correct for deviations of the compass which result from the magnetic influences of the aircraft structure and electrical system. Two screws on the face of the instrument are used to move the magnets and thus counterbalance the local magnetic influences acting on the main compass magnets. The two setscrews are labelled N-S and E-W.

Magnetic variation is the angular difference in degrees between the geographic north pole and the magnetic north pole. This variation is caused by the earth's magnetic field, which is constantly changing. Since variation differs according to geographic location, its effect on the compass cannot be removed by any type of compensation. Variation is called west variation when the earth's magnetic field draws the compass needle to the left of the geographic north pole and east variation when the needle is drawn to the right of the geographic north pole.

The compass needle is affected not only by the earth's magnetic field, but also by the magnetic fields generated when aircraft electrical equipment is operated, and by metal components in the aircraft. These magnetic disturbances within the aircraft, called deviation, deflect the compass needle from alignment with magnetic north.

To reduce this deviation, each compass in an aircraft is checked and compensated periodically by adjustment of the N-S and E-W magnets. The errors remaining after "swinging" the compass are recorded on a compass correction card mounted near the compass.

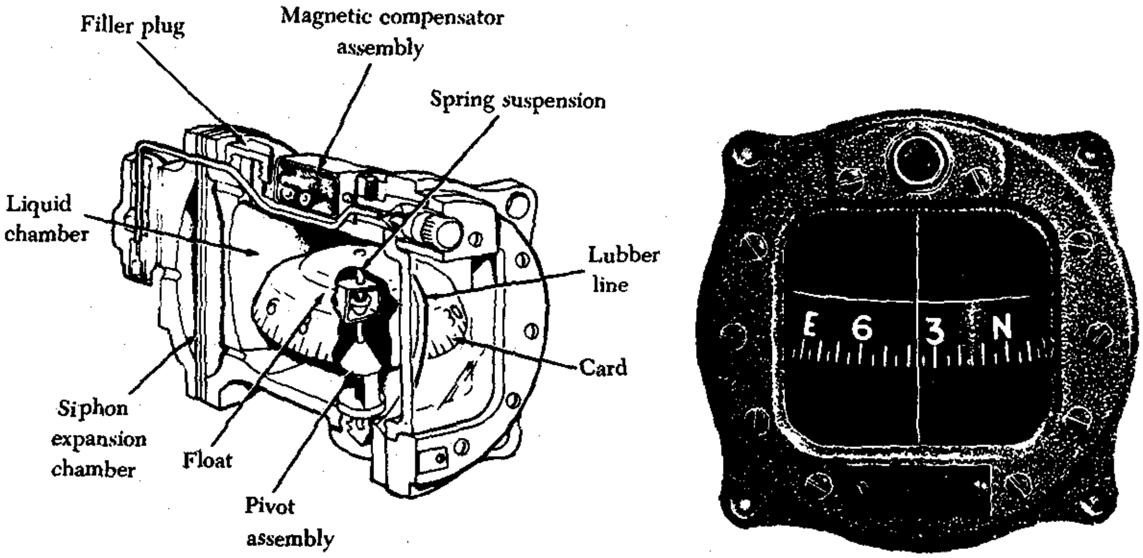


FIGURE 12-70. Magnetic compass.

The "swinging" (calibration) of a compass can be accomplished in flight or on the ground. Ground swinging of a compass is usually done with the aircraft at rest on a compass "rose." A compass rose (figure 12-71) is a circle laid out or painted on a level surface and graduated in degrees. The directions marked on the compass rose are magnetic directions, although true north is also marked on some compass roses.

Compass compensation procedures vary, depending on the type of aircraft. Requirements are often set up on a flight-hour and calendar basis. Most facilities perform compass checks anytime that equipment replacement, modification or relocation might cause compass deviation.

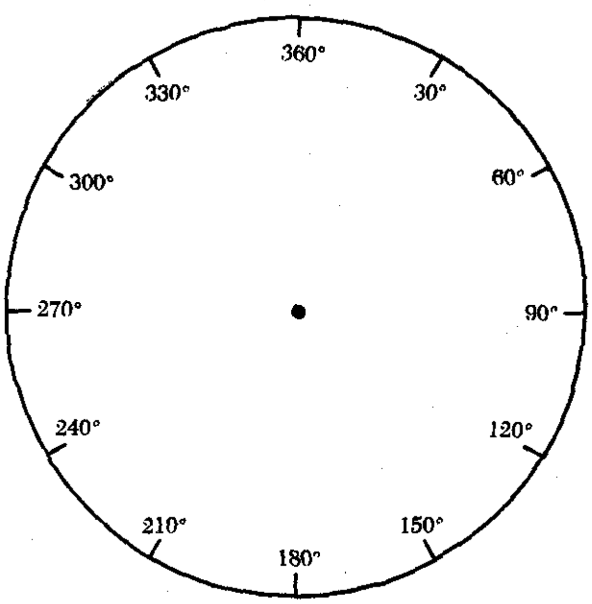


FIGURE 12-71. Typical compass rose.

An example of compass compensation is outlined in the following paragraphs. These procedures are general in nature and do not have specific application.

(1) The compensator should be set either to zero or in a position where it has no effect on the main compass magnets.

(2) The aircraft is placed directly on a south magnetic heading on the compass rose. The tail of tailwheel aircraft should be raised to level-flying position.

(3) Note the compass reading and record it. The deviation is the algebraic difference between the magnetic heading and the compass reading.

EXAMPLE:

On the south (180°) heading, the compass reading is 175.5°. This would be recorded as a deviation of +4.5° (180° — 175.5° = 4.5°). If the compass reading is too low, the deviation is plus; if the reading is too high, the deviation is minus.

(4) Align the aircraft on a magnetic north heading. Record the compass reading and compute the deviation.

EXAMPLE:

On the north (000°) heading, the compass reads 006.5°. Since the deviation is 6.5° too high, it is recorded as a minus deviation (—6.5°).

(5) The coefficient of north-south deviation is determined by subtracting, algebraically, the south deviation from the north deviation and dividing the remainder by 2:

$$\text{Coefficient} = \frac{(-6.5°) - (4.5°)}{2}$$

$$= \frac{-11°}{2}$$

$$= -5.5°$$

The coefficient of north-south deviation, which is the average of the deviation on the two headings, is —5.5°. The north-south compensator is adjusted by this amount, and the reading on the north heading will now be 001°. This adjustment also corrects the south deviation by the same amount, so that on a south heading the compass will now read 181°.

(6) Align the aircraft on a magnetic west (270°) heading on the compass rose. Record the compass reading and compute the deviation. Suppose the compass reads 276°, a deviation of —6°.

(7) Align the aircraft on a magnetic east (090°) heading. Record the compass reading and compute the deviation. Suppose the compass reading is exactly 90° on the magnetic east heading, a deviation of 0°.

(8) Compute the coefficient of east-west deviation:

$$\text{Coefficient} = \frac{0° - (-6°)}{2}$$

$$= \frac{+6°}{2}$$

$$= +3°$$

(9) While the aircraft is on the east heading, adjust the east-west compensator to add 3° to the compass reading. This reading then becomes 93° on the east heading and 273° on a west heading.

(10) Leaving the aircraft on an east magnetic heading, compute the coefficient of overall deviation. This coefficient is equal to the algebraic sum of the compass deviations on all four cardinal headings (north, east, south, and west), divided by 4:

$$\text{Coefficient} = \frac{(-6.5°) + 0° + 4.5° + (-6°)}{4}$$

$$= \frac{-8°}{4}$$

$$= -2°.$$

If the coefficient is greater than 1°, further compensation is usually accomplished. The compensation is not done with the magnetic compensation device. It is accomplished by re-aligning the compass, so that it is mounted parallel to the longitudinal axis of the aircraft.

(11) After the initial compensation is completed, the aircraft will be compensated again on headings of 30°, 60°, 120°, 150°, 210°, 240°, 300°, and 330°. The compass readings for each heading are recorded on a compass correction card. This card is then mounted as close as possible, to the instrument for ready reference. An example of a correction card is shown in figure 12–72.

The procedure described is a basic compensation procedure. Additional circuits around the compass rose should be made with the engine(s) and electrical and ratio equipment operating to verify the accuracy of the basic compensations.

Jacks, lifts, hoists, or any dolly needed to move and align the aircraft on the various headings of a compass rose should preferably be made of nonmagnetic material. When this is impossible, devices can be tested for their effect on the compass by moving them about the aircraft in a circle at the same distance that would separate them from the compass when they are being used. Equipment that causes a change in compass readings of more than one-quarter of a degree should not be used. Additionally, fuel trucks, tow tractors, or other aircraft containing magnetic metals should not be parked close enough to the compass rose to affect the compass of the aircraft being swung.

AIRCRAFT COMPASS

DATE..

	FOR	STEER
N	000°	000°
	030°	033°
	060°	060°
E	090°	095°
	120°	120°
	150°	149°
S	180°	175°
	210°	205°
	240°	334°
W	270°	265°
	300°	294°
	330°	326°

Calibrated by: ....

FIGURE 12-72. Compass correction card.

The magnetic compass is a simple instrument that does not require setting or a source of power. A minimum of maintenance is necessary, but the instrument is delicate and should be handled carefully during inspection. The following items are usually included in an inspection:

(1) The compass indicator should be checked for correct readings on various cardinal headings and re-compensated if necessary.

(2) Moving parts of the compass should work easily.

(3) The compass bowl should be correctly suspended on an anti-vibration device and should not touch any part of the metal container.

(4) The compass bowl should be filled with liquid. The liquid should not contain any bubbles nor have any discoloration.

(5) The scale should be readable and its illumination good.

AUTOPILOT SYSTEM

The automatic pilot is a system of automatic controls which holds the aircraft on any selected magnetic heading and returns the aircraft to that heading when it is displaced from it. The automatic pilot also keeps the aircraft stabilized around its horizontal and lateral axes.

The purpose of an automatic pilot system is primarily to reduce the work, strain, and fatigue of controlling the aircraft during long flights. To do this the automatic pilot system performs several functions. It allows the pilot to maneuver the aircraft with a minimum of manual operations. While under automatic control the aircraft can be made to climb, turn, and dive with small movements of the knobs on the autopilot controller.

Autopilot systems provide for one, two, or three axis control of the aircraft. Some autopilot systems control only the ailerons (one axis), others control ailerons and elevators or rudder (two axis). The three-axis system controls ailerons, elevators, and rudder.

All autopilot systems contain the same basic components: (1) Gyros, to sense what the airplane is doing; (2) servos, to move the control surfaces; and (3) an amplifier, to increase the strength of the gyro signals enough to operate the servos. A controller is also provided to allow manual control of the aircraft through the autopilot system.

Principle of Operation

The automatic pilot system flies the aircraft by using electrical signals developed in gyro-sensing units. These units are connected to flight instruments which indicate direction, rate-of-turn, bank, or pitch. If the flight attitude or magnetic heading is changed, electrical signals are developed in the gyros. These signals are used to control the operation of servo units which convert electrical energy into mechanical motion.

The servo is connected to the control surface and converts the electrical signals into mechanical force which moves the control surface in response to corrective signals or pilot commands. A basic autopilot system is shown in figure 12-73.

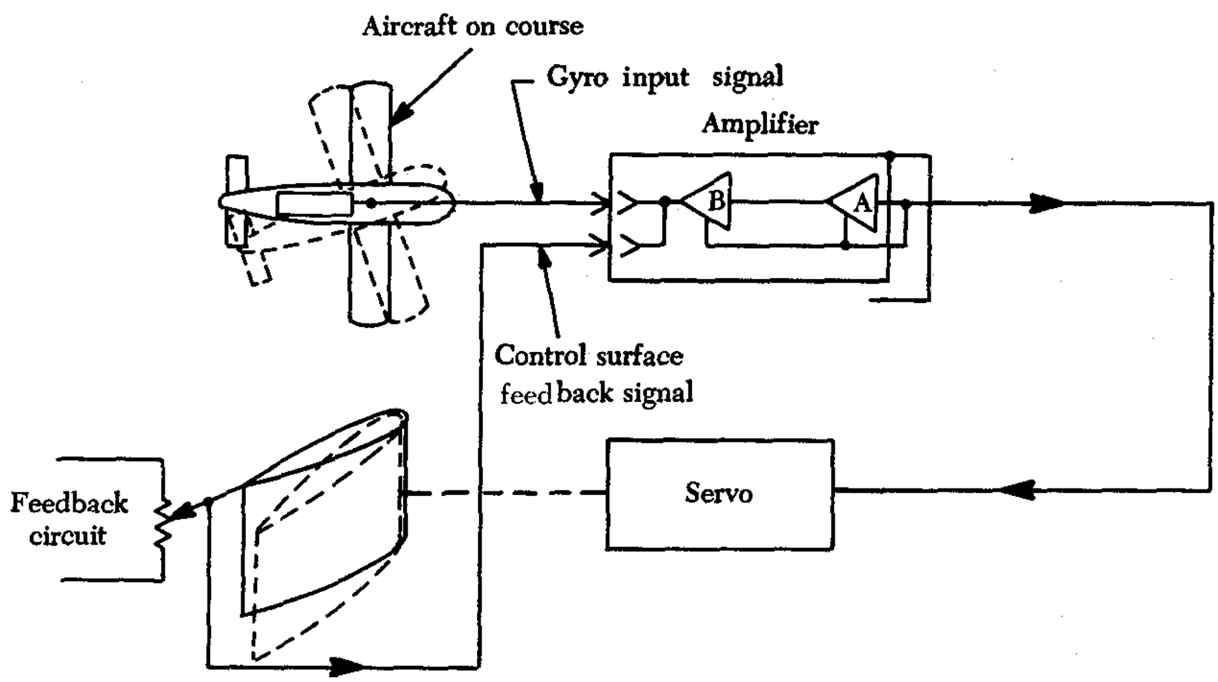


FIGURE 12–73. Basic autopilot system.

Most modern autopilots can be described in terms of their three major channels: (1) The rudder, (2) aileron, and (3) the elevator channels.

The rudder channel receives two signals that determine when and how much the rudder will move. The first signal is a course signal derived from a compass system. As long as the aircraft remains on the magnetic heading it was on when the autopilot was engaged, no signal will develop. But any deviation causes the compass system to send a signal to the rudder channel that is proportional to the angular displacement of the aircraft from the preset heading.

The second signal received by the rudder channel is the rate signal, which provides information anytime the aircraft is turning about the vertical axis. This information is provided by the turn-and-bank indicator gyro. When the aircraft attempts to turn off course, the rate gyro develops a signal proportional to the rate of turn, and the course gyro develops a signal proportional to the amount of displacement. The two signals are sent to the rudder channel of the amplifier, where they are combined and their strength is increased. The amplified signal is then sent to the rudder servo. The servo will turn the rudder in the proper direction to return the aircraft to the selected magnetic heading.

As the rudder surface moves, a followup signal is developed which opposes the input signal. When the two signals are equal in magnitude, the servo stops moving. As the aircraft arrives on course, the course signal will reach a zero value, and the rudder will be returned to the streamline position by the followup signal.

The aileron channel receives its input signal from a transmitter located in the gyro horizon indicator. Any movement of the aircraft about its longitudinal axis will cause the gyro-sensing unit to develop a signal to correct for the movement. This signal is amplified, phase-detected, and sent to the aileron servo which moves the aileron control surfaces to correct for the error.

As the aileron surfaces move, a followup signal builds up in opposition to the input signal. When the two signals are equal in magnitude, the servo stops moving. Since the ailerons are displaced from streamline, the aircraft will now start moving back toward level flight with the input signal becoming smaller and the followup signal driving the control surfaces back toward the streamline position. When the aircraft has returned to level flight in roll attitude, the input signal will again be zero. At the same time the control surfaces will be streamlined, and the followup signal will be zero.

The elevator channel circuits are similar to those of the aileron channel, with the exception that the

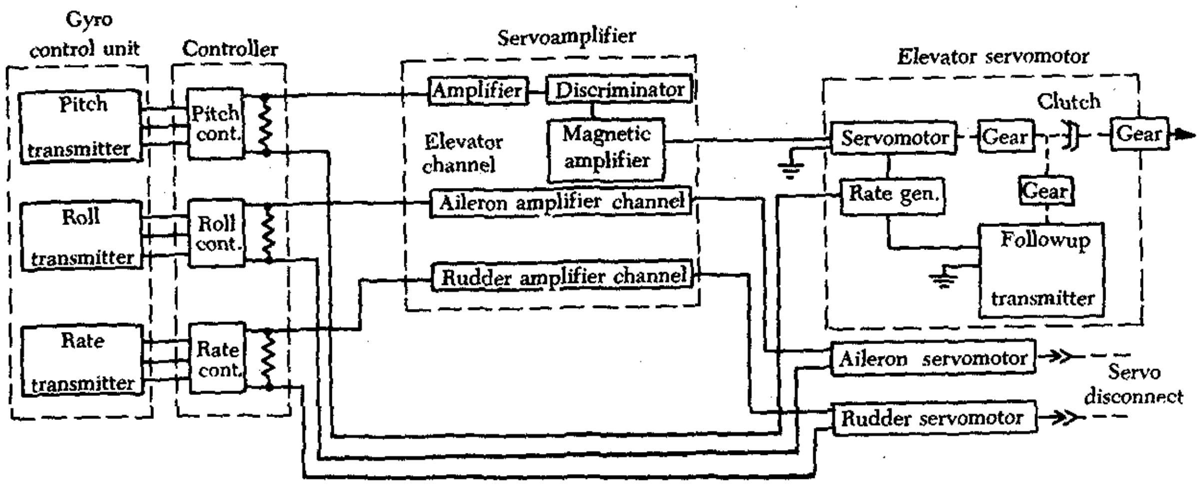


FIGURE 12–74. Autopilot block diagram.

elevator channel detects changes in pitch attitude of the aircraft. The circuitry of all three channels can be followed by referring to the block diagram in figure 12–74.

The foregoing autopilot system description was used to show the function of a simple autopilot. Most autopilots are far more sophisticated; however, many of the operating fundamentals are similar. Autopilot systems are capable of handling a variety of navigational inputs for automatic flight control.

BASIC AUTOPILOT COMPONENTS

The components of a typical autopilot system are illustrated in figure 12–75. Most systems consist of four basic types of units, plus various switches and auxiliary units. The four types of basic units are: (1) The sensing elements, (2) command elements, (3) output elements, and (4) the computing element.

Command Elements

The command unit (flight controller) is manually operated to generate signals which cause the aircraft to climb, dive, or perform coordinated turns. Additional command signals can be sent to the autopilot system by the aircraft's navigational equipment. The automatic pilot is engaged or disengaged electrically or mechanically, depending on system design.

While the automatic pilot system is engaged, the manual operation of the various knobs on the controller (figure 12–76) maneuvers the aircraft. By operating the pitch trim wheel, the aircraft can be made to climb or dive. By operating the turn knob, the aircraft can be banked in either direction. The engage switch is used to engage and disengage the autopilot. In addition, most systems have a disconnect switch located on the control wheel(s). This switch, operated by thumb pressure, can be used to disengage the autopilot system should a malfunction occur in the system.

One type of automatic pilot system has an engaging control that manually engages the clutch mechanism of the servomotor to the cable drum. A means of electrically disengaging the clutch is provided through a disconnect switch located on the control wheel(s).

Sensing Elements

The directional gyro, turn-and-bank gyro, attitude gyro, and altitude control are the sensing elements. These units sense the movements of the aircraft, and automatically generate signals to keep these movements under control.

Computer or Amplifier

The computing element consists of an amplifier or computer. The amplifier receives signals, determines what action the signals are calling for, and amplifies the signals received from the sensing elements. It passes these signals to the rudder, aileron, or elevator servos to drive the control surfaces to the position called for.

Output Elements

The output elements of an autopilot system are the servos which actuate the control surfaces. The

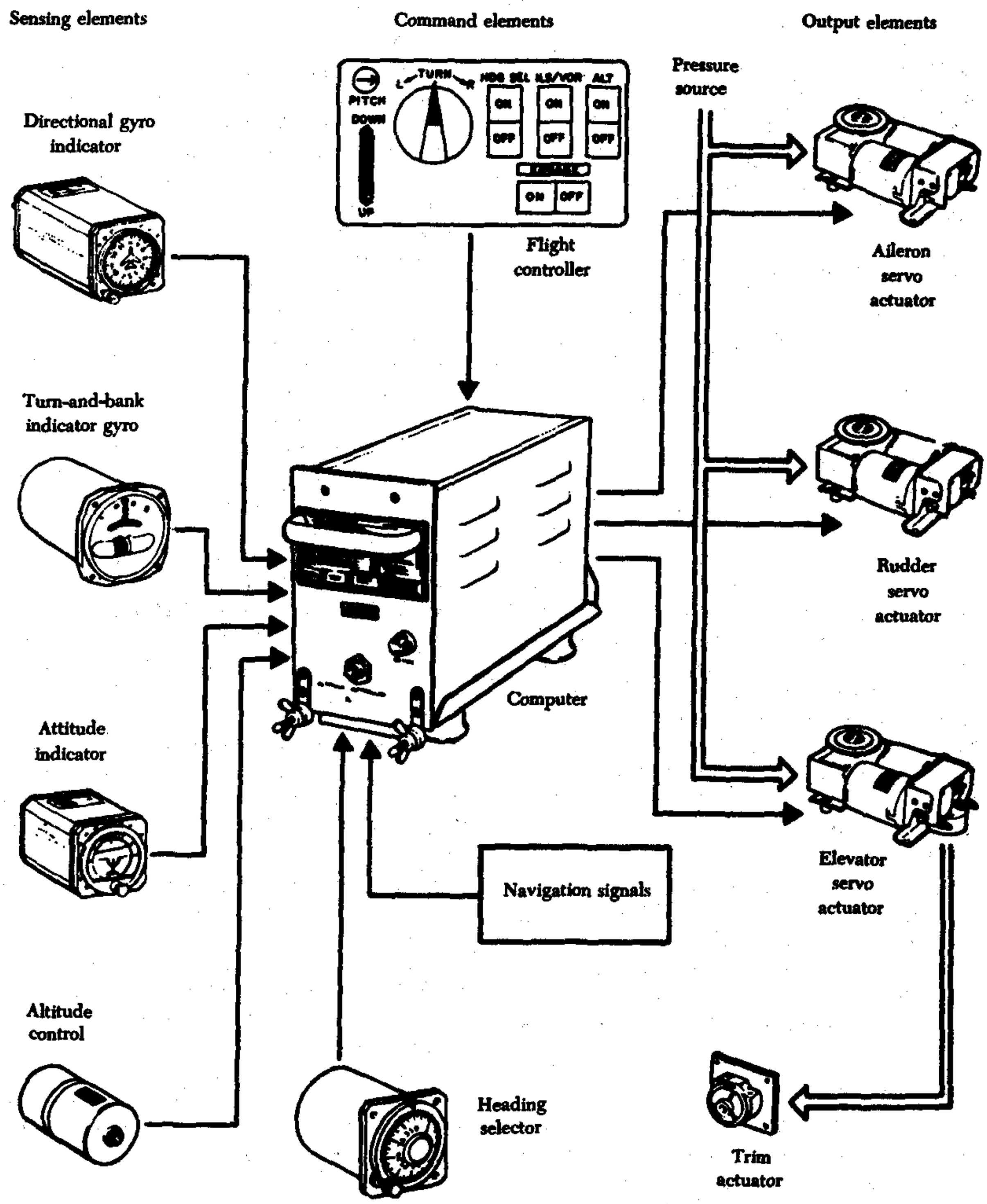


FIGURE 12-75. Typical autopilot system components.

majority of the servos in use today are either electric motors or electro/pneumatic servos.

An aircraft may have from one to three servos to operate the primary flight controls. One servo operates the ailerons, a second operates the rudder, and a third operates the elevators. Each servo drives its

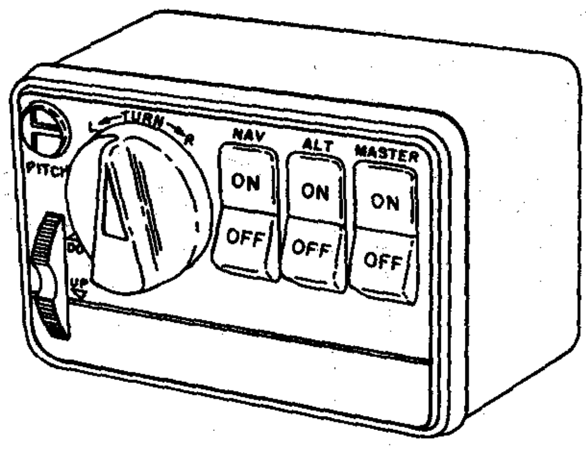


FIGURE 12-76. Typical autopilot controller.

associated control surface to follow the directions of the particular automatic pilot channel to which the servo is connected.

Two types of electric motor-operated servos are in general use. In one, a motor is connected to the servo output shaft through reduction gears. The motor starts, stops, and reverses direction in reponse to the commands of the gyros or controller. The other type of electric servo uses a constantly running motor geared to the output shaft through two magnetic clutches. The clutches are arranged so that energizing one clutch transmits motor torque to turn the output shaft in one direction; energizing the other clutch turns the shaft in the opposite direction.

The electro/pneumatic servos are controlled by electrical signals from the autopilot amplifier and actuated by an appropriate air pressure source. The source may be a vacuum system pump or turbine engine bleed air. Each servo consists of an electro/magnetic valve assembly and an output linkage assembly.

FLIGHT DIRECTOR SYSTEMS

A flight director system is an instrument system consisting of electronic components that will compute and indicate the aircraft attitude required to attain and maintain a preselected flight condition. "Command" indicators on the instrument indicate how much and in what direction the attitude of the aircraft must be changed to achieve the desired result. The computed command indications relieve the operator of many of the mental calculations required for instrument flights, such as interception angles, wind drift correction, and rates of climb and descent.

A flight director system has several components; the principal ones are the gyroscope, computer, and the cockpit presentaton.The gyro detects deviations from a preselected aircraft attitude. Any force exerted against the gyroscope is electrically transmitted to the computer, which in turn, sends a computed signal to the flight indicator, telling the operator what must be done with the controls. When using a flight director system, the operator is, in a sense, acting as a servo, following orders given by the command indicators.

The computers used in the various types of flight director systems are basically the same; however, the numbers and types of functions available will vary between systems because of the mission of a particular aircraft, the limited aircraft space available for installation, and the excessive cost of functions not absolutely required.

The instrument panel presentations and operating methods vary considerably between different systems. Command indications may be presented by several different symbols, such as bar-type command indicators with different types of movements, a phantom aircraft symbol, or two-element crossbar indicators.

Many flight director systems are equipped with an "altitude-hold" function, which permits selection of a desired altitude; the flight director computes the pitch attitude necessary to maintain this particular altitude.

A flight director greatly simplifies problems of aerial navigation. Selection of the VOR function electronically links the computer to the omnirange receiver. After selection of the desired omnicourse, the flight director will direct the bank attitude necessary to intercept and maintain this course.

Flight director systems are designed to offer the greatest assistance during the instrument approach phase of flight. ILS localizer and glide slope signals are transmitted through the receivers to the computer, and are presented as command indications. With the altitude-hold function, level flight can be maintained during the maneuvering and procedure turn phase of the approach. Once inbound on the localizer, the command signals of the flight director are maintained in a centered or zero condition.

Compensation for wind drift is automatic. Interception of the glide slope will cause a downward indication of the command pitch indicator. Any deviation from the proper glide slope path will cause a fly-up or fly-down indication on the flight director pitch command symbol. When altitude-hold is being used, it automatically disengages when the glide slope has been intercepted.

A flight director system not only shows the present situation, but also predicts the future consequences of this situation. For example, a momentary change in attitude is detected by the computer, and command symbol movement is made to correct this condition possibly before an altitude error can result. Thus, greater precision is achieved with less mental effort on the part of the aircraft operator.

AUTOPILOT SYSTEM MAINTENANCE

The information in this section does not apply to any particular autopilot system, but gives general information which relates to all autopilot systems. Maintenance of an autopilot system consists of visual inspection, replacement of components, cleaning, lubrication, and an operational checkout of the system.

With the autopilot disengaged, the flight controls should function smoothly. The resistance offered by the autopilot servos should not affect the control of the aircraft. The interconnecting mechanism between the autopilot system and the flight control system should be correctly aligned and smooth in operation. When applicable, the operating cables should be checked for tension.

An operational check is important to assure that every circuit is functioning properly. An autopilot operational check should be performed on new installations, after replacement of an autopilot component, or whenever a malfunction in the autopilot system is suspected.

After the aircraft's main power switch has been turned on, allow the gyros to come up to speed and the amplifier to warm up before engaging the autopilot. Some systems are designed with safeguards that prevent premature autopilot engagement.

While holding the control column in the normal flight position, engage the system using the engaging control (switch, handle).

After the system is engaged, perform the operational checks specified for the particular aircraft. In general, the checks consist of:

(1) Rotate the turn knob to the left; the left rudder pedal should move forward, and the control column wheel should move to and the control column wheel should move slightly aft.

(2) Rotate the turn knob to the right; the right rudder pedal should move forward, and the control column wheel should move to the right. The control column should move slightly aft. Return the turn knob to the center position; the flight controls should return to the level-flight position.

(3) Rotate the pitch-trim knob forward; the control column should move forward.

(4) Rotate the pitch-trim knob aft; the control column should move aft.

If the aircraft has a pitch-trim system installed, it should function to add downtrim as the control column moves forward, and add uptrim as the column moves aft. Many pitch-trim systems have an automatic and a manual mode of operation. The above action will occur only in the automatic mode.

Check to see if it is possible to manually override or overpower the autopilot system in all control positions. Center all the controls when the operational checks have been completed.

Disengage the autopilot system and check for freedom of the control surfaces by moving the control columns and rudder pedals. Then re-engage the system and check the emergency disconnect release circuit. The autopilot should disengae each time the release button is actuated.

When performing maintenance and operational checks on a specific autopilot system, always follow the procedure recommended by the aircraft or equipment manufacturer.

Annunciator System

Instruments are installed for two purposes, one to display current conditions, the other to notify of unsatisfactory conditions. Colored scales are used; usually green for satisfactory; yellow for caution or borderline conditions; red, for unsatisfactory conditions. As aircraft have become more complex with many systems to be monitored, the need for a centralized warning system became apparent. The necessity to coordinate engine and flight controls emphasized this need. What evolved is an annunciator or master warning system (figure 12–77).

Certain system failures are immediately indicated on an annunciator panel on the main instrument panel. A master caution light and a light indicating the faulting system flash on. The master light may be reset to "Off," but the indicating light will remain "On" until the fault is corrected or the equipment concerned is shut down. By resetting, the master caution light is ready to warn of a sub-

SYSTEM	ATA NUMBER	INDICATION
Aircraft Fuel	2800	Fuel Pressure Low
Engine Fuel	7300	Fuel Pressure Low
Electrical	2400	Inverter Out
Generator	2400	Generator Out
Generator	2400	Generator Overheated
Starting	8000	Starter Engaged
Engine Oil	7900	Oil Pressure Low
Landing Gear	3200	Brake Pressure Low
Landing Gear	3200	Not Locked Down
Landing Gear	3200	Anti-Skid Out
Air Conditioning	2100	Cabin Pressure High
Air Conditioning	2100	Cabin Pressure Low
Flight Control	2700	Speed Brake Extended
Stabilizer	5500	Not Set for Takeoff
Engine Exhaust	7800	Thrust Reversal Pressure Low
Aux Power	4900	APU Exhaust Door Not Open
Doors	5200	Cabin Door Unlocked
Doors	5200	Cargo Door Unlocked
Navigation	3400	Mach Trim Computer Out
Electrical	2400	Normal Bus Tie Open
Auto Flight	2200	Auto Pilot Off
Hydraulic	2900	Hydraulic Pressure Low
Firewarning	2600	AFT Compartment Overheated

FIGURE 12–77. Warning in annunciator system.

sequent fault even before correction of the initial fault. A press to test light is available for testing the circuits in this system.

One late model business jet has the sensing devices divided into groups, according to their method of operation. The fast group responds to heat and uses bimetallic strips set at predetermined temperatures. The second group responds to pressure changes and uses a flexible chamber that moves when pressurized. The third group consists of mechanically operated switches and/or contacts on a relay.

An annunciator system may include any or all of the following indications or others as applicable.

Aural Warning System

Aircraft with retractable landing gear use an aural warning system to alert the crew to an unsafe condition. A bell will sound if the throttle is retracted and the landing gear is not in a down and locked condition (figure 12–78).

Aural warning systems range in complexity from the simple one just described to that system necessary for safe operation of the most complex transport aircraft.

A typical transport aircraft has an aural warning system which will alert the pilot with audio signals to: An abnormal takeoff condition, landing condition, pressurization condition, mach-speed condition, an engine or wheel well fire, calls from the crew call system, and calls from the secal system. Shown in figure 12–78 are some of the problems which trigger warning signals in the aural warning system. For example: a continuous horn sounding during landing would indicate the landing gear is not down and locked when flaps are less than full up and the throttle is retarded. The corrective action would be to raise the flaps and advance the throttle.

(See figure 12–78 on next page)

STAGE OF OPERATION	WARNING SYSTEM	WARNING SIGNAL	CAUSE OF WARNING SIGNAL ACTIVATION	CORRECTIVE ACTION
Landing	Landing gear ATA 3200	Continuous horn	Landing gear is not down and locked when flaps are less than full up and throttle is retarded to idle.	Raise flaps Advance throttle
In flight	Mach warning ATA 3400	Clacker	Equivalent airspeed or mach number exceeds limits.	Decrease speed of aircraft
Takoff	Flight control ATA 2700 Aux power ATA 4900	Intermittent horn	Throttles are advanced and any of following conditions exist. 1. Speed brakes are not down 2. Flaps are not in takeoff range 3. Auxiliary power exhaust door is open 4. Stabilizer is not in the takeoff setting.	Correct the aircraft to proper takeoff conditions.
Inflight	Pressurization ATA 2100	Intermittent horn	If cabin pressure becomes equal to atmospheric pressure at the specific altitude (altitude at time of occurrence).	Correct the condition.
Any stage	Fire warning ATA 2600	Continuous bell	Any overheat condition or fire in any engine or nacelle, or main wheel or nose wheel well, APU engine or any compartment having firewarning system installed. Also whenever the firewarning system is tested.	1. Lower the heat in the area where-in the F/W was activated. 2. Signal may be silenced by pushing the F/W bell cut-out switch or the APU cutout switch.
Any stage	Communications ATA 2300	High chime	Any time captain's call button is pressed at external power panel forward or rearward cabin attendant's panel	Release button or if button remains locked in, pull button out.
Any stage	Communications secal system* ATA 2300	Tow tone hi-low chime or single low chime.	Whenever a signal has been received by an HF or VHF communication system and decoded by the secal* decoder.	Press reset button on secal system control panel.

*NOTE: Secal system is the Selective Calling System: Each aircraft is assigned a particular four tone audio combination for identification purposes. A ground station will key the signal whenever contact with that particular aircraft is desired. The signal will be decoded by the airborne secal decoder and the crew allerted by the secal warning system.

FIGURE 12-78. Aural warning system.